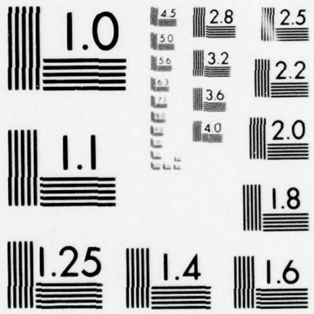
AD-A077 373 UNITED TECHNOLOGIES CORP STRATFORD CT SIKORSKY AIRCR--ETC F/G 15/5
ADVANCED STRUCTURES CONCEPTS R AND M/COST ASSESSMENTS.(U)
SEP 79 T N COOK, B F KAY DAAJ02-77-C-0061
UNCLASSIFIED SER-510016 USARTL-TR-79-18 NL

1 OF 3
AD
A077373



MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS-1963-A

USARTL-TR-79-18

(12)

5C

LEVEL II

ADVANCED STRUCTURES CONCEPTS R&M/COST ASSESSMENTS

Thomas N. Cook, Bruce F. Kay
Sikorsky Aircraft Division
United Technologies Corporation
Stratford, Conn. 06602

September 1979

Final Report for Period October 1977 - February 1979

Approved for public release;
distribution unlimited.

Prepared for

APPLIED TECHNOLOGY LABORATORY
U. S. ARMY RESEARCH AND TECHNOLOGY LABORATORIES (AVRADCOM)
Fort Eustis, Va. 23604

APPLIED TECHNOLOGY LABORATORY POSITION STATEMENT

This report is the result of a contractual effort by the Reliability, Maintainability and Mission Technology Technical Area, Aeronautical Systems Division of the Applied Technology Laboratory, US Army Research and Technology Laboratories (AVRADCOM), to investigate the R&M/cost characteristics of advanced structures concepts proposed or under development for Army helicopters.

The objectives of this contractual effort were to investigate the R&M/cost characteristics of advanced structures concepts, to establish a methodology for considering R&M in selecting advanced structures design concepts/arrangements, and to impose R&M as a design issue in advanced structures concepts. Additionally, using the R&M design analysis techniques and the life-cycle cost assessment procedures developed under this effort, preliminary advanced structures concepts R&M design guidelines were to be prepared.

This report has been reviewed and the R&M analysis technique and cost assessment methodology presented are considered to be reasonable and acceptable approaches for improving the R&M characteristics of advanced structures concepts proposed or under development for Army helicopters. Further efforts are planned to apply the R&M/cost assessment techniques developed under this effort to planned Army helicopter advanced structures programs.

The technical monitor for this contractual effort was Mr. Thomas E. Condon of the Aeronautical Systems Division, Reliability, Maintainability and Mission Technology Technical Area.

Unclassified
SECURITY CLASSIFICATION OF THIS PAGE (When Data Entered)

19 REPORT DOCUMENTATION PAGE

READ INSTRUCTIONS BEFORE COMPLETING FORM

18 1. REPORT NUMBER: USARTL-TR-79-18

2. GOVT ACCESSION NO.

3. RECIPIENT'S CATALOG NUMBER: 9

6 4. TITLE (and Subtitle): ADVANCED STRUCTURES CONCEPTS R&M/COST ASSESSMENTS

5. TYPE OF REPORT & PERIOD COVERED: Final Report
Oct[redacted] 1977 — Febr[redacted] 1979

6. PERFORMING ORG. REPORT NUMBER: 14 SER-510016

7. AUTHOR(s): 10
Thomas N. Cook
Bruce F. Kay

8. CONTRACT OR GRANT NUMBER(s): 15 DAAJ02-77-C-0061

9. PERFORMING ORGANIZATION NAME AND ADDRESS
Sikorsky Aircraft Division
United Technologies Corporation
Stratford, Connecticut 06602

10. PROGRAM ELEMENT, PROJECT, TASK AREA & WORK UNIT NUMBERS: 16
62209A IL262209AH76
00 215 EK

11. CONTROLLING OFFICE NAME AND ADDRESS
Applied Technology Laboratory, U. S. Army
Research and Technology Laboratories (AVRADCOM)
Fort Eustis, Virginia 23604

12. REPORT DATE: 11 Septe~~mber~~ 1979 17 00

13. NUMBER OF PAGES: 239

14. MONITORING AGENCY NAME & ADDRESS (if different from Controlling Office)

15. SECURITY CLASS. (of this report): Unclassified 12 237

15a. DECLASSIFICATION/DOWNGRADING SCHEDULE

16. DISTRIBUTION STATEMENT (of this Report)

Approved for public release; distribution unlimited.

17. DISTRIBUTION STATEMENT (of the abstract entered in Block 20, if different from Report)

18. SUPPLEMENTARY NOTES

19. KEY WORDS (Continue on reverse side if necessary and identify by block number)

Reliability	Cost	Structures
Maintainability	Helicopter	Composites
Maintenance	Aircraft	Composite Materials
Repair	Airframe	Advanced Composites
Damage Tolerance	Fuselage	Materials Testing

20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

Recent programs have investigated various aspects of the design and manufacture of advanced composite airframe structures for helicopters. Evaluation of the reliability and maintainability (R&M) and operating cost characteristics of the evolving design concepts has been limited, however. The objective of this program has been to assess the overall potential of advanced composite structures from the standpoint of R&M and life-cycle cost.

DD FORM 1 JAN 73 1473 EDITION OF 1 NOV 65 IS OBSOLETE
S/N 0102-014-6601

Unclassified
SECURITY CLASSIFICATION OF THIS PAGE (When Data Entered)

323 800 JOB

A survey was made of in-service experience with helicopter airframe structures, concentrating particularly on bonded structures and composite materials. The surveys included visits to Army helicopter depots where typical types of damage were examined and discussed. A review was also made of published data on composites in use with fixed-wing aircraft. It was established that with the exception of some secondary structure, experience with composites in helicopter airframe applications is very limited, and that quantitative reliability factors cannot yet be established. However, it was concluded that the majority of failures with advanced composites will occur from external causes, primarily as a result of damage by impact.

Reliability and maintainability factors in composite structures design were identified and defined. The principal damage modes were related to the environmental hazards encountered by the Army helicopter and to the level of exposure of generic airframe structures to these hazards. The damage tolerance of various composite materials was rated with respect to specific types of damage, and design characteristics having the potential for mitigating impact damage were identified.

Inspection and repair of primary structure were determined to be critical issues with respect to the maintainability of advanced composites. It was concluded that methods being developed for use in the fixed-wing community will be largely incompatible with the Army field environment and that further work is needed to develop suitable techniques. Repair of large area battle damage may be very difficult, and a modular design approach is suggested as one of the possible solutions to this problem.

Laboratory testing was conducted to assess the damage tolerance of several composites in both monolithic and sandwich construction, and to test the effectiveness of simple field-type repairs. Varying degrees of impact tolerance were demonstrated, and simple patching techniques were found to be effective for some types of damage. Subsurface damage in honeycomb sandwich panels was found in some cases to cause a significant structural weakening of the panel without producing visual evidence of damage.

A method was developed to assess and rank the R&M characteristics of advanced composite structures designs. The method involves a systematic evaluation of the many variables affecting the reliability and maintainability of helicopter airframe structures. Damage potential, damage tolerance, repairability and replaceability are among the key factors evaluated. The product of this analysis is a qualitative rating of structural reliability, hardware reliability and maintainability.

A method was developed to assess the life-cycle cost potential of advanced structures designs. The method uses the results of the R&M assessment and historical cost data on present-day helicopter airframes to arrive at a life-cycle cost projection for a composite design. Sensitivity studies indicate that advanced composites can be cost effective and that the life-cycle costs are dominated by the initial cost of manufacture and by the structural reliability of the design.

The R&M and cost characteristics of four advanced composites concepts, representing a cross-section of helicopter airframe structures, were analyzed in detail. Preliminary R&M design guidelines for advanced composite structures for Army helicopters were established.

PREFACE

This program of advanced structures concepts R&M and cost assessments was performed by the Sikorsky Aircraft Division of United Technologies Corporation under Contract DAAJ02-77-C-0061 for the Applied Technology Laboratory (ATL), U.S. Army Research and Technology Laboratories (AVRADCOM), Fort Eustis, Virginia. The program was conducted under the technical direction of Mr. Thomas E. Condon of the Reliability, Maintainability and Mission Technology Technical Area of ATL.

The authors wish to acknowledge contributions to this program made by the following Sikorsky Aircraft personnel. Mr. George Mardoian of the Structures and Materials Branch supervised the damage tolerance and repairability testing of composite materials and assisted with analysis of the test results. Mr. Thomas Harman of the Airframe Design Section was primarily responsible for definition and analysis of the advanced structures concepts. Mr. Richard Corbeille of the Reliability and Maintainability Section assisted with development of the life-cycle cost estimating methodology.

Appreciation is also extended to personnel at the U.S. Army Depots, Corpus Christi, Texas and New Cumberland, Pennsylvania for their advice and assistance.

TABLE OF CONTENTS

TABLE OF CONTENTS (Continued)

LIST OF ILLUSTRATIONS

LIST OF ILLUSTRATIONS (Continued)

LIST OF ILLUSTRATIONS (Continued)

LIST OF ILLUSTRATIONS (Continued)

LIST OF ILLUSTRATIONS (Continued)

LIST OF TABLES

LIST OF TABLES (Continued)

LIST OF TABLES (Continued)

LIST OF TABLES (Continued)

INTRODUCTION

The field of advanced composite materials has witnessed remarkable growth over the last few years. Until recently, applications of composite materials to aircraft were almost exclusively in the nature of fiberglass fairings and minor secondary structures. While fiberglass and secondary structural uses still predominate aircraft applications, advanced composite materials are now being used in a variety of new areas, including the design of primary structure and major dynamic components. Development work with advanced composites is expanding enormously, and airframes constructed entirely from these new materials are now receiving serious study.

Advanced composites offer a number of attractions to the aircraft designer. They combine high strength with low weight and they are adaptable to a variety of manufacturing processes. Because they lend themselves to monolithic types of construction, composites eliminate many assembly details, reduce complexity and lower manufacturing costs. In many areas composites have greater damage tolerance and are more survivable against combat damage than metals.

An aspect of advanced composites design receiving relatively little attention thus far is the one to which this program is addressed: reliability and maintainability (R&M) and its associated life-cycle costs. There is little question that advanced composite structures of almost any conceivable type can be constructed, and that these structures can be designed to possess the required strength for aircraft use. There is some uncertainty about the suitability of these structures for many of the environments in which they might be placed, however,particularly environments as hostile and austere as those in which the Army helicopter operates.

The uncertainty connected with the R&M of advanced composite structures stems in large part from the limited study the subject has received. This program is an important step toward a better understanding of the R&M and life-cycle cost implications of advanced composite structures for Army helicopters.

COMPOSITE MATERIALS AND FORMS OF CONSTRUCTION

COMPOSITE MATERIALS

Advanced composite materials used in airframe structures include fiber reinforced plastics, sandwich materials and adhesives. Described below are generic types of composites considered appropriate for airframe construction.

Fiber Reinforced Plastics

The fiber reinforced plastics are thermosetting materials containing a matrix of epoxy resin and fibers. The fiber materials are fiberglass, Kevlar, graphite and boron. The distinctive characteristics of each material are described below. Typical mechanical properties are listed in Table 1.

TABLE 1. TYPICAL PROPERTIES OF COMPOSITES AND ALUMINUM

Material	Ultimate Tension (psi)	Ultimate Compression (psi)	Tensile Modulus (psi X 10^6)	Density (lb/in^3)
Aluminum (2024 T3)	60,000	37,000*	10.5	.100
Fiberglass/Epoxy (181 Style Fabric; Warp Direction)	48,C00	50,000	3.4	.067
Fiberglass/Epoxy (Unidirectional E-Glass; 0° Layup)	160,000	90,000	5.5	.067
Kevlar/Epoxy (181 Style Fabric, Warp Direction)	56,000	24,000	4.5	.048
Graphite/Epoxy (0° Layup)	160,000	160,000	17.	.055
Boron/Epoxy (0° Layup)	192,000	360,000	30.	.073

*Yield Strength

Fiberglass/Epoxy

This composite material has been used in the construction of aircraft longer than any other. In the common woven form, fiberglass/epoxy is a relatively low-strength, low-modulus material whose use has been relegated to nonstructural applications such as fairings, cowlings and doors. It is the lowest cost composite, and because of the long-term experience with the material, manufacturing and repair techniques for fiberglass are well developed.

Fiberglass is also available in unidirectional form known as E-glass and S-glass. These materials have high strength-to-weight ratios and excellent impact characteristics, and have been used in such structural applications as cargo floors and helicopter rotor blades.

Kevlar/Epoxy

Kevlar/epoxy is a relatively new aramid fiber material produced by the DuPont Company. It has a lower density than fiberglass although its tensile strength is high. A very low compression strength restricts its use in applications where compression loads are significant. Kevlar is straw-colored as opposed to the translucent quality of fiberglass, which makes it slightly more difficult to laminate. It also requires special techniques for drilling and trimming. Conventional metalworking tools produce ragged edges when used on Kevlar.

Graphite/Epoxy

Graphite/epoxy is a high-strength/high-modulus material that is used for primary structure applications. It is also a less ductile material which makes it very susceptible to stress concentration effects and impact damage.

Graphite/epoxy is expensive; prices currently range between $40 and $80 per pound. However, due to expanded use and increased production, the cost per pound has been dropping and is expected to continue to do so in the foreseeable future.

Boron/Epoxy

Boron/epoxy is a high-strength material with an extremely high modulus of elasticity. It is most useful in structures designed for stiffness or compression strength. The cost of boron/epoxy is much higher than that of other composite materials including graphite. Boron/epoxy has very large diameter fibers that are stiff flexurally; therefore, the material can only be formed to gentle contours. In addition, drilling and trimming can only be accomplished with special tools of the diamond-coated variety.

Hybrids

Hybrid composite designs are achieved by combining various materials. In such designs, materials are combined in a manner that takes advantage of their unique individual properties. Combining Kevlar skins with graphite stiffeners, for example, produces a structure with the shear and impact strength of Kevlar and the compression strength of graphite. Through the use of hybrids, structures can be optimized for particular conditions.

SANDWICH CORE MATERIALS

Honeycomb and foam are the basic types of filler material used for sandwich structure. Typical properties of these materials are listed in Table 2.

TABLE 2. TYPICAL PROPERTIES OF CORE MATERIALS

Material	Density (lb/ft^3)	Compressive Strength (psi)	Shear Strength (psi)	Compression Modulus (psi)
Syntactic Foam	35	6,000	1,800	26,000
Polyurethane Foam	8	210	84	5,600
Aluminum Honeycomb	6	680	455	240,000
Nomex Honeycomb	6	825	260	60,000

Honeycomb

Honeycomb is a hexagonal-shaped cellular material made from metal foil or plastic sheet. Both varieties are procured in slabs of the required thickness and then machined to final form or shape.

Aluminum honeycomb has the highest shear strength of the standard honeycomb materials. The plastic honeycombs have lower strength but are more resistant to corrosion. The most common plastic honeycomb used in aircraft applications is Nomex. This product is DuPont's nylon-fiber paper treated with a heat-resistant phenolic resin. It is not as strong as aluminum honeycomb but offers greater resiliency, making it attractive for applications where impact may be experienced.

Foams

Foams are very lightweight, low-strength materials that are normally precast to shape and used as fillers to separate and provide form to facing material during lamination. They are also used to provide a stabilizing effect against buckling in certain applications.

Polyurethane foam is typical of the nonstructural foams. It is a two-part foam made by mixing premeasured amounts of two chemicals that form a gaseous reaction. This operation must be carefully controlled to produce consistent quality.

For structural forms, heavier syntactic foams are used. These foams contain glass micro-balloons (small hollow glass spheres) mixed in an epoxy matrix. Also used to inhibit crushing of lightweight core materials, the syntactic foams are more easily produced than the two-part gaseous foams and can be cast in place in some applications.

ADHESIVES

Two generic forms of structural adhesive are used: paste and film. The film adhesives are made in thin sheets that are placed between the surfaces to be bonded, brought into intimate contact by pressure or clamps and heated to effect a cure. The film adhesives offer maximum bond strength but require special preparations. These include maintaining strict cleanliness of faying surfaces, precise fitting to insure bond line uniformity and accurate control of temperature for curing.

Paste adhesives generally have lower strength than film adhesive but are able to accommodate a certain amount of variation between faying surfaces. Room-temperature-cure adhesives are available but have poorer properties than the hot-bond adhesives.

MATERIAL FORMS

Composite structure is formed from laminated plies of composite material. The individual plies are made from fabric or unidirectional tape.

Fabric

Fabric material is made by weaving strands of fibers in a mutually perpendicular pattern to form a cloth (Figure 1). Structural properties of the fabric are orthotropic, with the highest strenth and stiffness in a direction parallel to the fibers. This requires that fabric be properly oriented with respect to the applied loads (Figure 2).

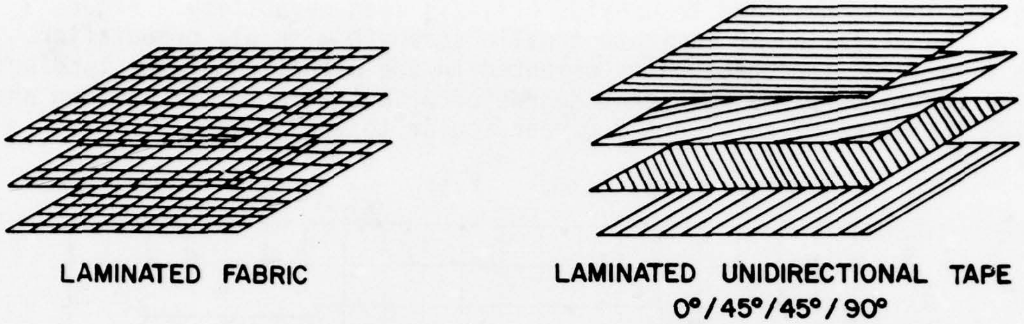


Figure 1. Material Forms

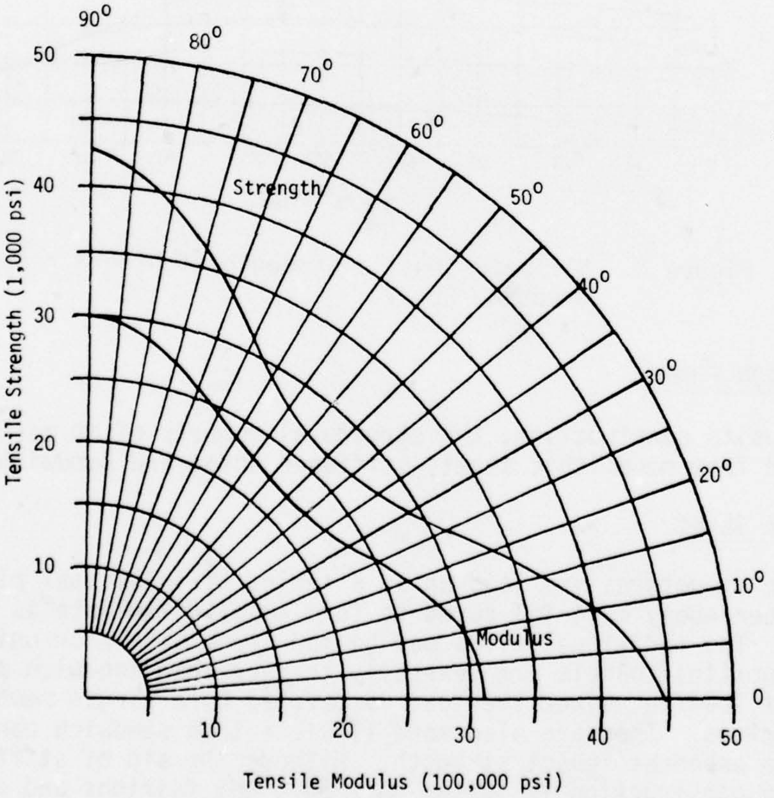


Figure 2. Directional Properties in Tension of Parallel-Laminated 181 Glass-Fabric Laminate Made With Epoxy Resin (MIL-R-9300)

Unidirectional Tape

Unidirectional tape consists of collimated fibers in a resin matrix (Figure 1). Extremely high strength properties are obtained parallel to the fibers, but the material is very weak in a direction perpendicular to the fibers. As a result, laminates constructed from unidirectional composites are cross-plied to provide off-axis load capability. Figure 3 shows the variation in laminate tensile strength with ply orientation. In the figure, 0° indicates plies oriented in the direction of the loading, ± 45° indicates plies with that degree of orientation to the loading and ± 90° indicates plies oriented perpendicular to the loading.

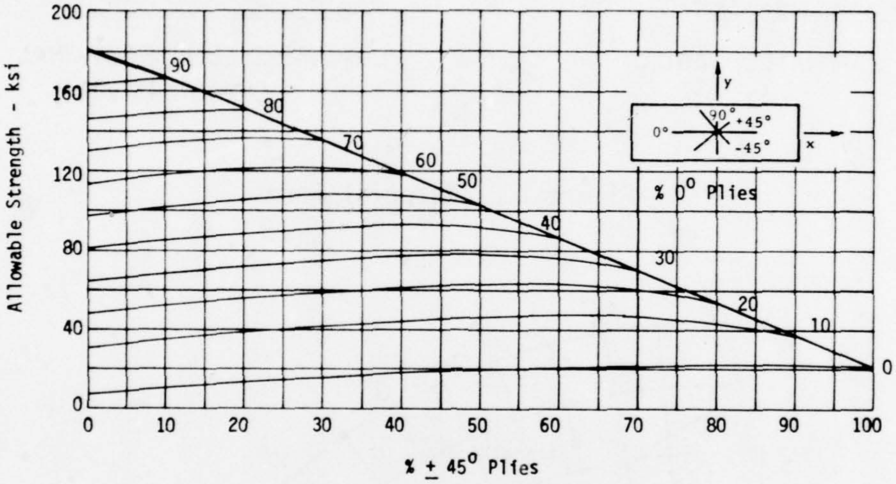


Figure 3. Ultimate Tensile Strength of High-Strength Graphite/Epoxy

CONSTRUCTION FORMS

With composite construction, the structural members of an airframe are fabricated from monolithic sheet, stiffened sheet and sandwich panels.

Monolithic Sheet

Monolithic structures are laid up as a series of individual plies of reinforced fiber/epoxy material cured to form a solid laminate as shown in Figure 4. The individual plies may be made from fabric or unidirectional tape. Monolithic panels are generally thicker than sandwich panels for equivalent loading, since the load is carried by a single member instead of two facings. They are also more flexible than sandwich panels and this resilience enhances impact strength. Without the aid of stiffeners, monolithic construction is suited only to light fairings and covers.

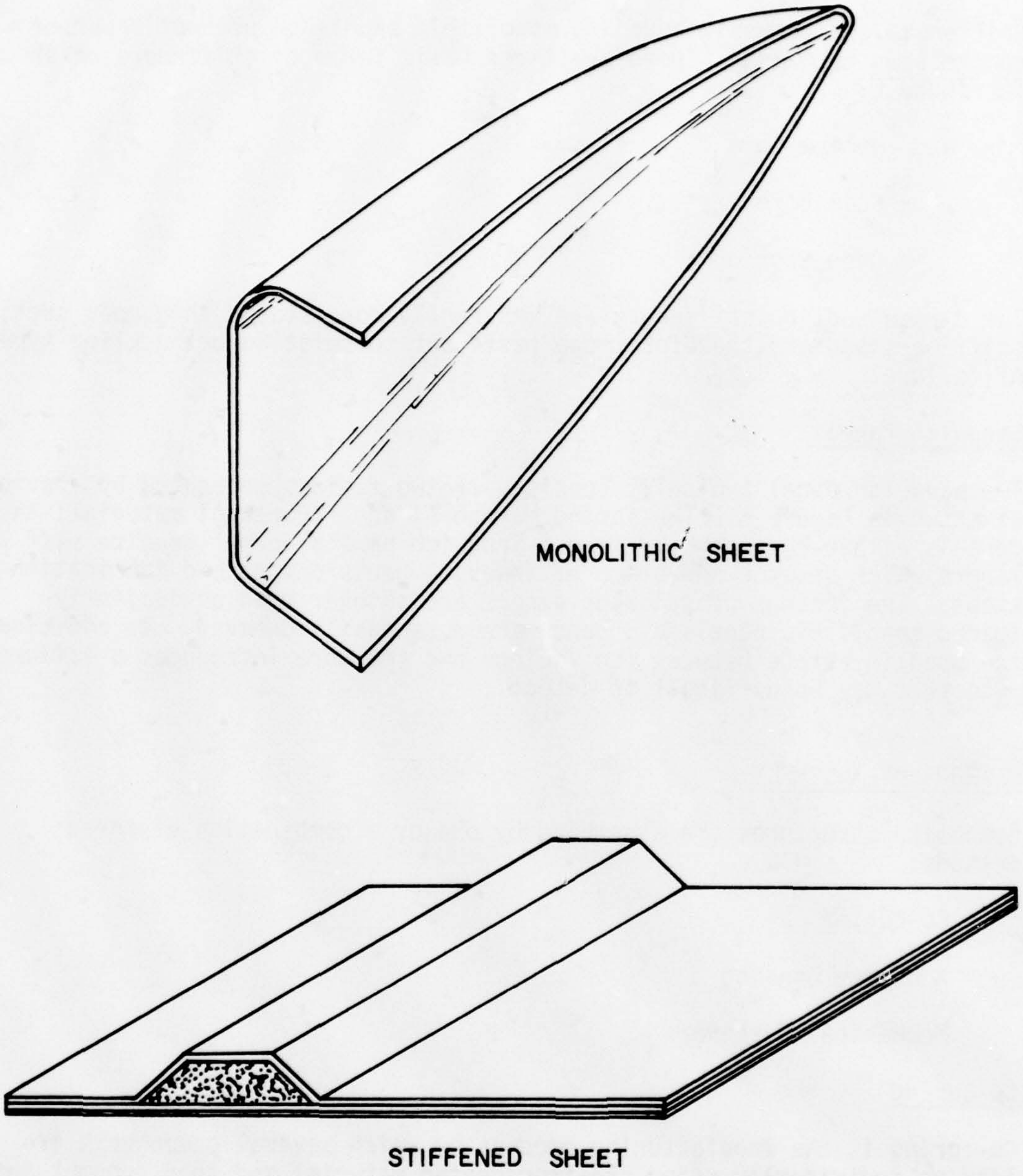


Figure 4. Forms of Sheet Construction

Stiffened Sheet

Stiffeners are commonly added to monolithic panels to prevent shear or compression buckling. There are three basic forms of stiffeners as shown in Figure 5.

Hollow Core

Foam Core

Open Section

The closed section stiffeners are inherently more stable than open section stiffeners and are therefore more resistant to twisting or buckling types of failure.

Sandwich Panels

The sandwich panel typically consists of two facings separated by a core as shown in Figure 6. The facing may be of any structural material; the core is either honeycomb or foam. Sandwich panels do not require stiffeners, which has the advantage of lowering parts counts and fabrication steps. The facings of sandwich panels are thinner than equivalently loaded monolithic panels and hence are more easily damaged. In addition, the bond interface between the facings and the core introduces a failure mode that may be difficult to detect.

METHODS OF ASSEMBLY

Composite structures are assembled by one or a combination of three methods:

Co-Curing

Adhesive Bonding

Mechanical Fasteners

Co-Curing

Co-curing is the manufacturing process by which several components are laid up individually using pre-impregnated material and then brought into intimate contact under pressure and cured together as an integral assembly. The disadvantage of co-curing is that the joined components are inseparable for practical purposes, and must be cut apart when replacement is necessary.

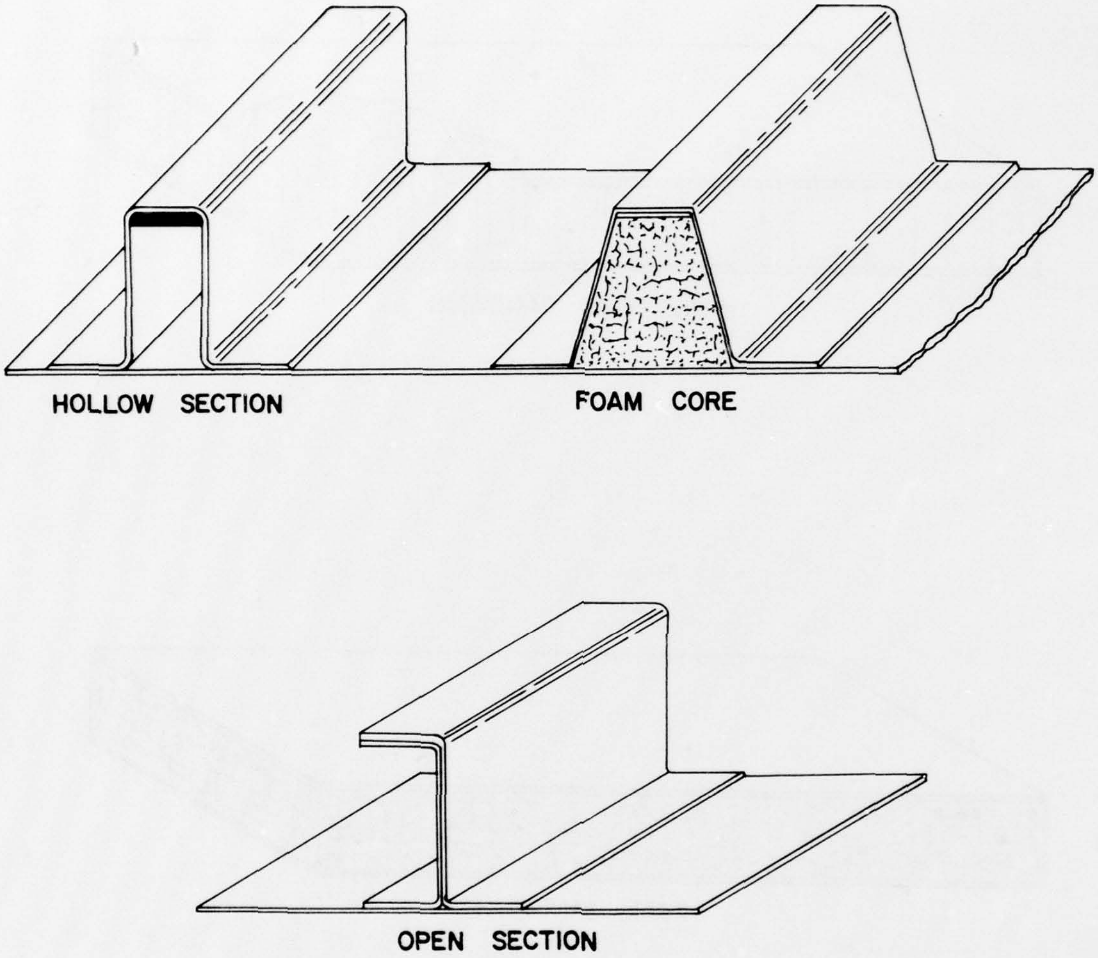
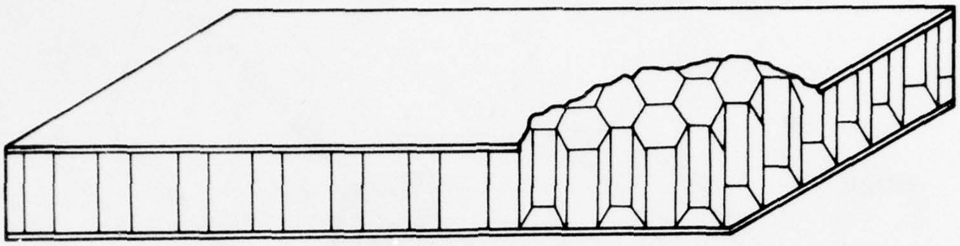
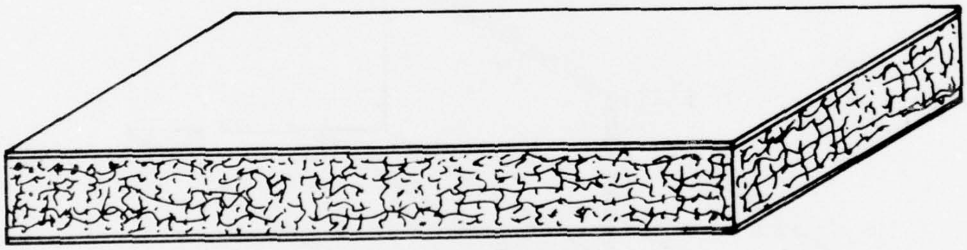


Figure 5. Stiffener Forms



HONEYCOMB SANDWICH



FOAM SANDWICH

Figure 6. Forms of Sandwich Construction

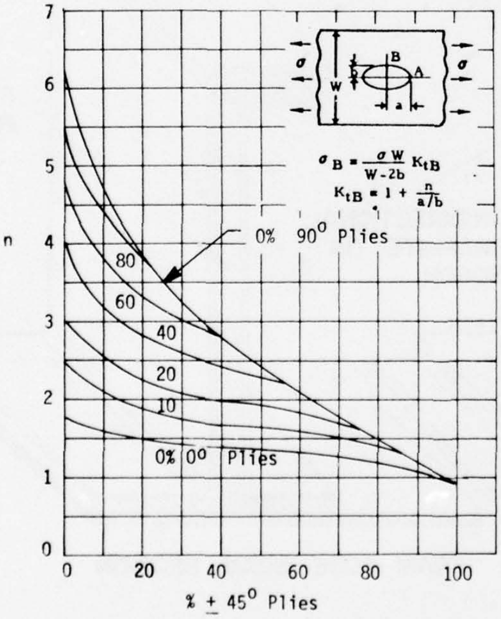
Adhesive Bonding

Adhesive bonding joins two previously cured structural components using paste or film adhesives. The bond is usually cured under heat and pressure. A clean simple structural joint is produced; but like co-curing, adhesive joints are considered permanent and present a handicap for repair or replacement. Adhesive bonding also requires absolute cleanliness during preparation of the joints, and verification of joint strength may be difficult under field conditions. In addition, certain adhesives require refrigeration and have limited shelf lives.

Mechanical Fasteners

Under certain conditions, composite structures may be joined mechanically using rivets or bolts. However, the nonyielding characteristics of composite materials place restrictions on this method of assembly. Whenever a hole is drilled in composite material, stress concentrations develop, even under static loading conditions.

For example, Figure 7 shows that the stress concentration factor for a hole drilled in unidirectional graphite/epoxy is approximately seven. That is, the stress in a drilled structure is increased approximately 700% over that in an undrilled structure. Mechanical fasteners cannot be placed in composite structures unless the structure has been designed to accommodate them. Special considerations must be given to the layup of local reinforcements where mechanical fasteners are used.



Note:

σ_B = Stress at point B

σ = Gross stress on element

Figure 7. Stress Concentration Factor at Point B, High-Strength Graphite/Epoxy

BASIC STRUCTURAL COMPONENTS

The airframe of a helicopter is comprised of structural components of three basic types:

Skins and Webs

Frames, Beams, Bulkheads and Longerons

Structural Fittings

Skins and Webs

The skins, together with bulkhead, frame and beam webs, are the principal members supporting shear loads in a structure. These members may be of either monolithic or sandwich construction.

Frames, Beams, Bulkheads and Longerons

Frames, beams, bulkheads and longerons are the primary structural members supporting tension, compression and bending loads in the structure. These components may be fabricated in a variety of open-section and closed-section shapes using monolithic, stiffened sheet or sandwich construction. Figures 8, 9 and 10 show several typical designs.

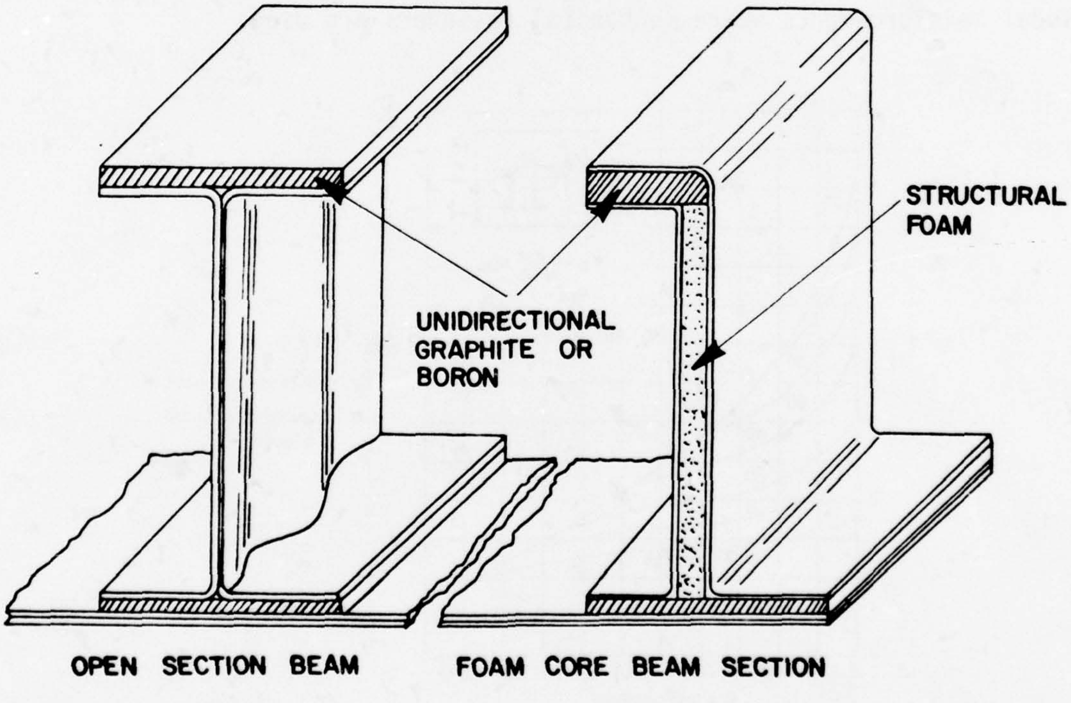
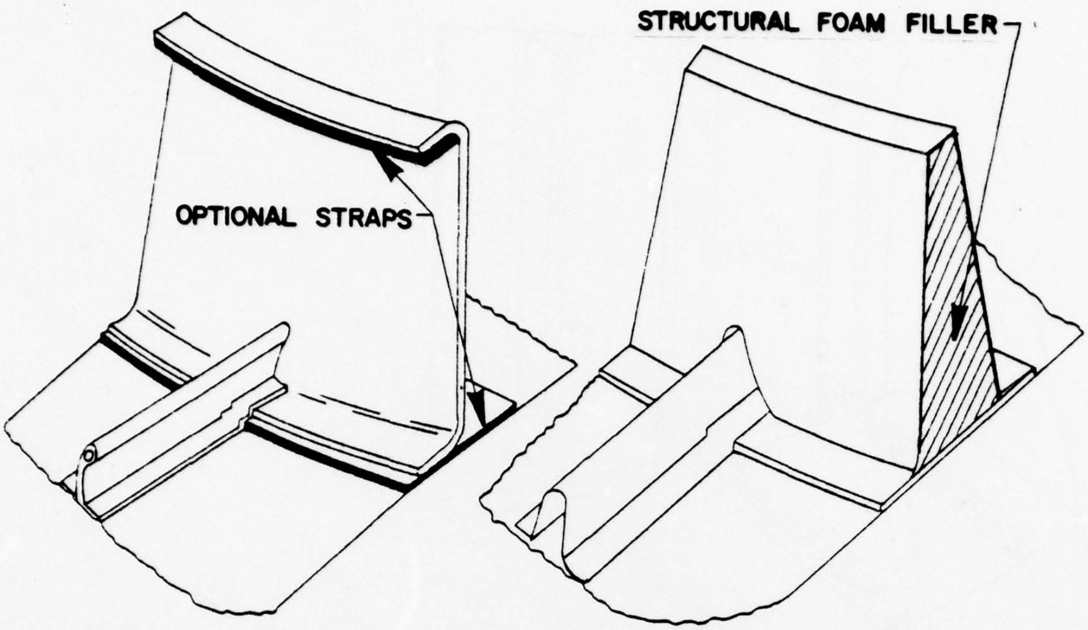
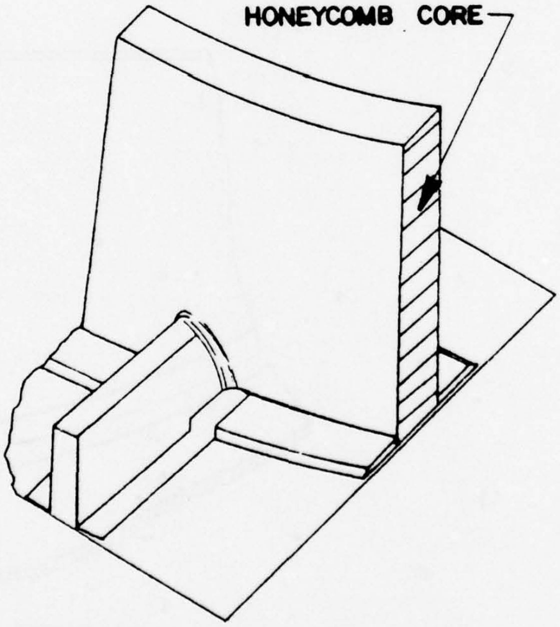


Figure 8. Typical Beam Sections



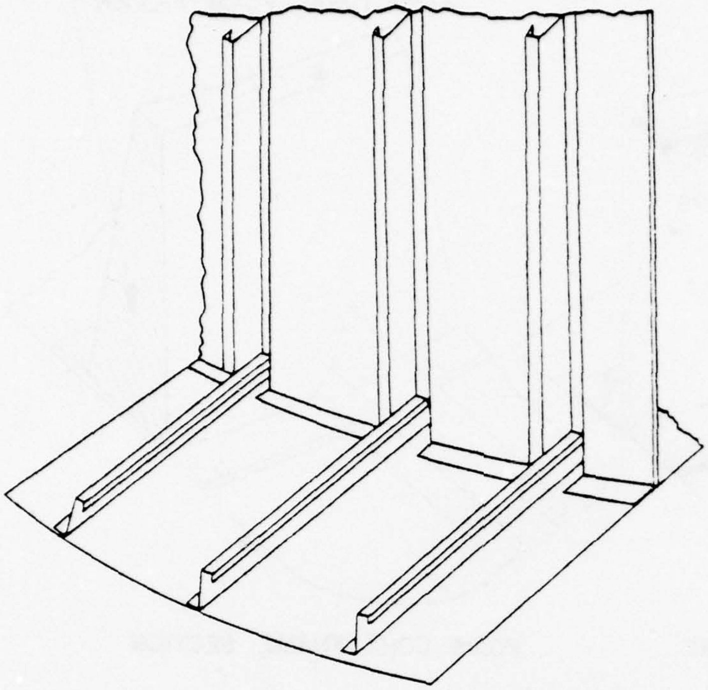
OPEN SECTION FRAME

FOAM CORE FRAME SECTION

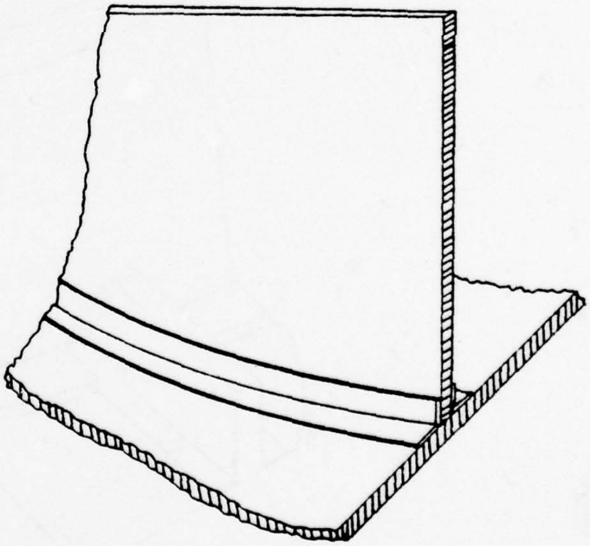


HONEYCOMB CORE FRAME SECTION

Figure 9. Typical Frame Sections



STIFFENED WEB BULKHEAD

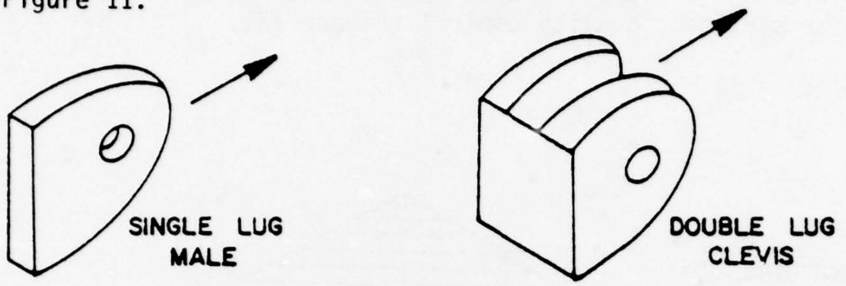


SANDWICH BULKHEAD

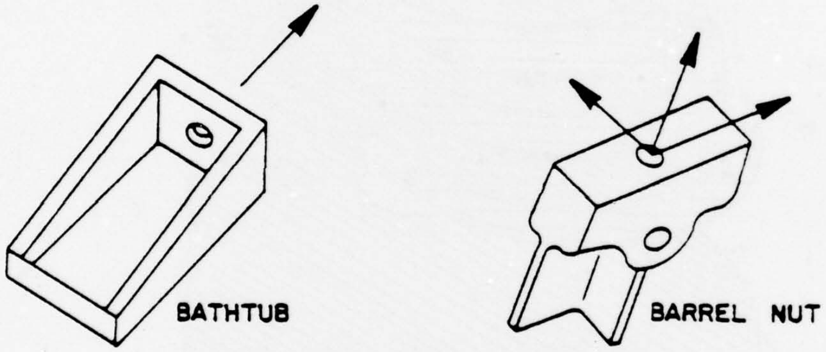
Figure 10. Typical Bulkhead Designs

Structural Fittings

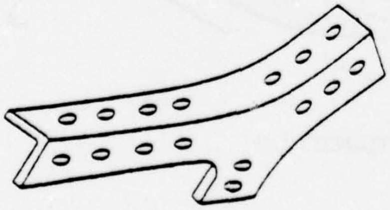
The introduction of concentrated loads and the presence of highly loaded regions in a structure create the need for structural fittings. These fittings are frequently characterized by three-dimensional loading as shown in Figure 11.



SINGLE BOLT SHEAR FITTINGS



SINGLE BOLT TENSION FITTINGS



MULTIPLE BOLT SHEAR FITTING

Figure 11. Three Basic Types of Concentrated Load Introduction Fittings

In a conventional metal structure, structural fittings are machined from metal forgings, bars or plate stock. Because of the complexity involved in producing efficient and economical fittings from composites, composite structures frequently retain metal fittings at selected locations. Where composite fittings are used, they are fabricated using rigidly defined layups under stringent quality control (Figure 12).

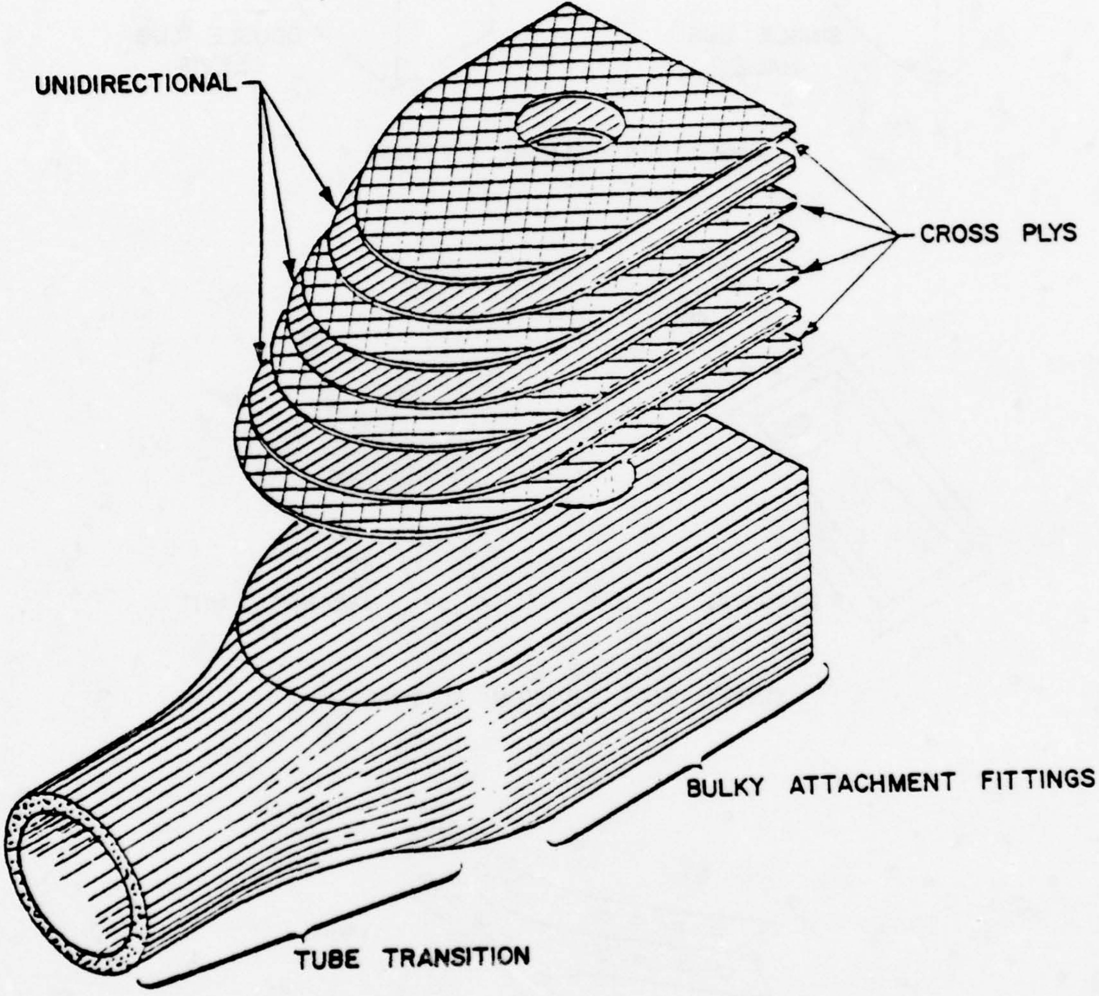


Figure 12. Composite Fitting

ADVANCED STRUCTURES DESIGN CONCEPTS

One of the requirements of this program was to develop a technique for assessing the R&M characteristics and R&M related life-cycle costs of advanced composite structures concepts. Preparatory to this task, helicopter airframe structures were classified and described generically, and candidate composite designs were selected for analysis.

A typical utility class helicopter was selected as the model for this study. The aircraft fuselage is defined in terms of primary structures and secondary structures as follows:

Primary Structures

Cockpit Canopy
Cockpit Lower Structure
Upper Fuselage Structure
Lower Fuselage Structure
Rear Fuselage
Tail Cone
Tail Rotor Pylon
Stabilator

Secondary Structures

Floors
Fairings and Cowlings
Aircraft Doors

In the following pages, each of these structures is defined in terms of its general configuration and the structural criteria governing its design. Following the generic description of each structure, conceptual designs are presented. Two composite designs and a metal baseline design are presented for each type of structure. To facilitate comparisons between concepts, a tabular format has been used, and only the predominant characteristics of each design have been listed.

The composite structures concepts were taken primarily from the two studies on advanced structural designs for utility class helicopters (References 1 and 2). In cases where the references did not provide sufficient information on a design, Sikorsky's Airframe Design Group further developed the

---

[1] Hoffstedt, P.J., and Swatton, S., ADVANCED HELICOPTER STRUCTURAL DESIGN INVESTIGATION, Boeing Vertol Company, USAAMRDL-TR-75-56A, U.S. Army Air Mobility Research and Development Laboratory, Fort Eustis, VA, March 1976, AD A024662.

[2] Rich, M. J., INVESTIGATION OF ADVANCED HELICOPTER STRUCTURAL DESIGNS, Sikorsky Aircraft Division, USAAMRDL-TR-75-59A, U.S. Army Air Mobility Research and Development Laboratory, Fort Eustis, VA, May 1976, AD 026246.

concept or synthesized a concept from two or more sources. Composite structures designs have also been drawn from existing aircraft of Sikorsky manufacture, and in the case of the transmission support structure, from Reference 3 report. All of the metal baseline designs have been taken from existing Sikorsky helicopter models. The remainder of this section of the report describes the selected design concepts.

PRIMARY STRUCTURE

Cockpit Canopy

Cockpit canopy structures are characterized by a gridwork of posts and sills that in conjuntion with the windshields and windows forms a transparent enclosure for the helicopter flight crew (Figure 13 and Table 3). The structures are designed primarily for aerodynamic pressure loading and do not contribute significantly to fuselage bending strength against primary flight inertia loads. Composite structures concepts are described in Table 4.

Cockpit Lower Structure

The lower cockpit is the primary structural support for the nose section (Figure 14 and Table 5). Critical design loads are usually derived from inertia forces acting on the crew, equipment and structural mass. Typically, the cockpit is constructed as a semi-monocoque structure cantilevered from the mid-fuselage.

Interface constraints are complex because of high density packaging of flight controls, avionics, electrical equipment, etc., although the mounting provisions for these installations require only minimal reinforcement. The exception is the support structure adjacent to seats, nose landing gear or gun turrets which requires heavy reinforcement.

In addition to interface constraints, the nose section must be crashworthy. This is accomplished by providing structural elements that are capable of absorbing crash impact energies. Currently, energy-absorbing structures are either ductile aluminum or honeycomb. Because composites lack energy-absorbing properties, for the near term lower cockpit structures are expected to contain significant quantities of metallic structure. Composite structures concepts are described in Table 6.

Upper Fuselage Structure

Upper fuselage structures are of semi-monocoque construction with frames and beams at load introduction points (Figure 15 and Table 7). The most heavily loaded members are the lift system, landing gear, weapon pylons,

---

[3] Kay, B. F., Lowry, D. S., and Rich, M.J., STUDY TO INVESTIGATE DESIGN, FABRICATION AND TEST OF LOW COST CONCEPTS FOR LARGE HYBRID HELICOPTER FUSELAGE - PHASE III, Sikorsky Aircraft Division, NASA Contractor Report 158988, National Aeronautics and Space Administration, Hampton, Va., February 1979.

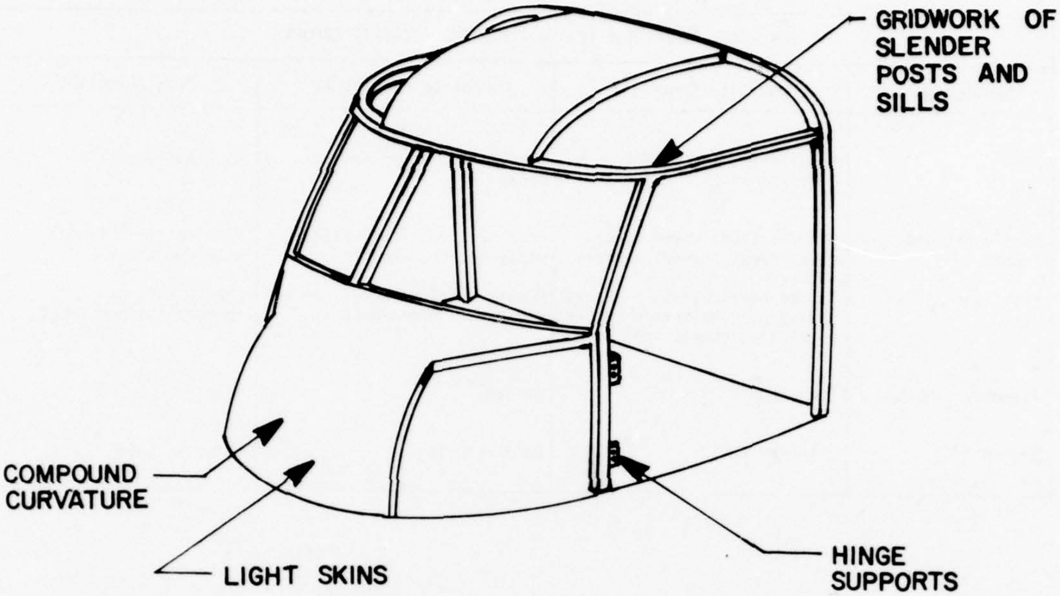


Figure 13. Cockpit Canopy

TABLE 3. COCKPIT CANOPY STRUCTURE ATTRIBUTES	
Attribute	**Description**
Contour	Predominantly compound curvature
Accessibility	Generally good, except for restrictions imposed by equipment
Interfaces and Concentrated Loads	Bracketry-type supports required for mounting controls, hatches, etc. Normally consist of local reinforcements on structure or minor machined fittings. Relatively large number required per aircraft.
Special Constraints	Windshield posts must be slender to minimize interference with visibility.
Load Intensity	Light, allowing minimum gage construction.

TABLE 4. STRUCTURAL DESIGN CONCEPTS - COCKPIT CANOPY

Sub-Component	Composite Concept I	Composite Concept II	Metal Baseline
Skin	Monolithic; woven fiberglass/epoxy	Monolithic; woven Kevlar/epoxy	Aluminum sheet
Stiffeners and Frames	Open section; woven fiberglass/epoxy channel members	Open section; woven Kevlar/epoxy channel members	Open section; formed aluminum sheet
Posts and Sills	Closed section; woven fiberglass/epoxy hatshaped members with polyurethane foam cores	Closed section; woven Kevlar/epoxy hat-shaped members	Closed section; formed aluminum sheet
Assembly Method	Co-cured	Bonded	Riveted
Source	Sikorsky CH-53	Reference (2)	Sikorsky S-61

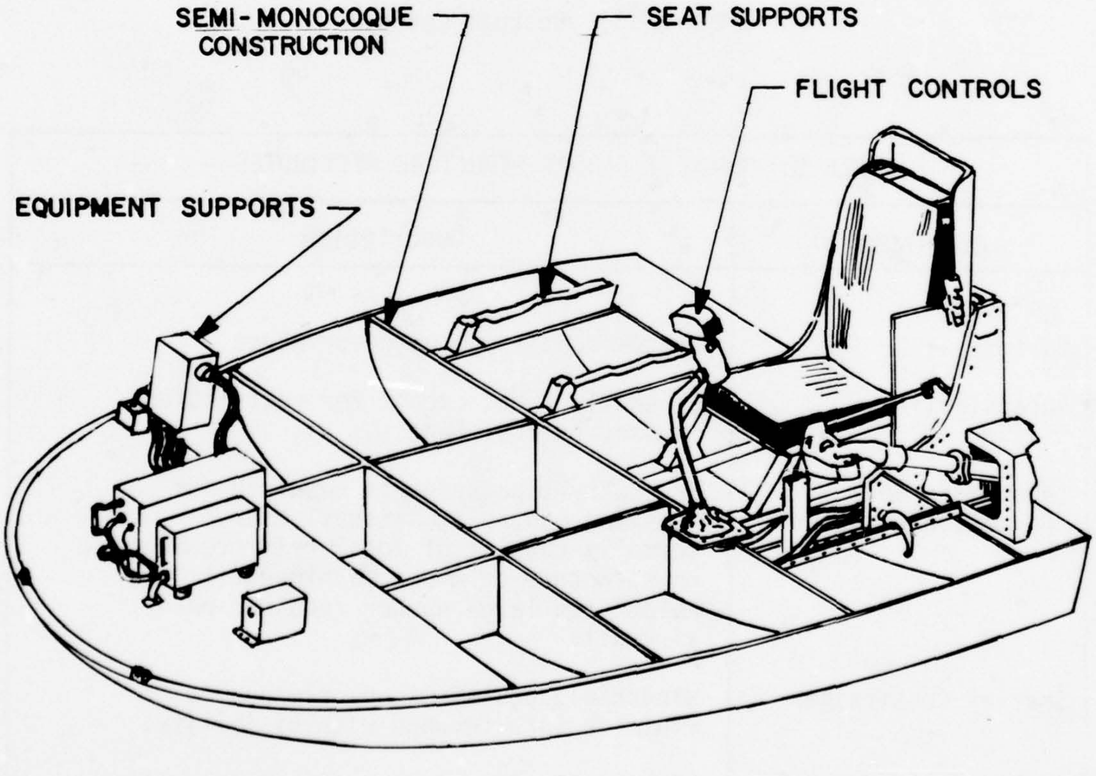


Figure 14. Cockpit Lower Structure

TABLE 5. LOWER COCKPIT STRUCTURE ATTRIBUTES

Attribute	Description
Contour	**Predominantly compound curvature**
Accessibility	Poor because of high density packaging of systems
Interfaces and Concentrated Loads	Bracketry-type supports required for mounting equipment. Heavy reinforcements required for landing gears, seats, gun turrets.
Load Intensities	Light to moderate, except heavy adjacent to concentrated load interfaces
Special Constraints	Structure must be capable of attenuating crash impact energies

TABLE 6. STRUCTURAL DESIGN CONCEPTS - LOWER COCKPIT

Sub-Component	Composite Concept I	Composite Concept II	Metal Baseline
Skin	Monolithic; woven Kevlar/epoxy	Sandwich; woven Kevlar/epoxy facings with aluminum honeycomb core	Aluminum sheet
Stiffeners	Closed section; cross-plied graphite/epoxy hat-shaped members with polyurethane foam cores	None	Open section; aluminum members channel
Frames and Beams	Sandwich; cross-plied graphite/epoxy facings and caps with Nomex honeycomb core	Closed section; woven Kevlar/epoxy channel members. Integrated with sandwich skins to serve as edge closeouts	Open sections; built-up channels and I-beams fabricated from aluminum sheet and extrusions
Assembly Method	Riveted	Co-cured	Riveted
Source	Reference (2)	Reference (1)	Sikorsky UH-60A

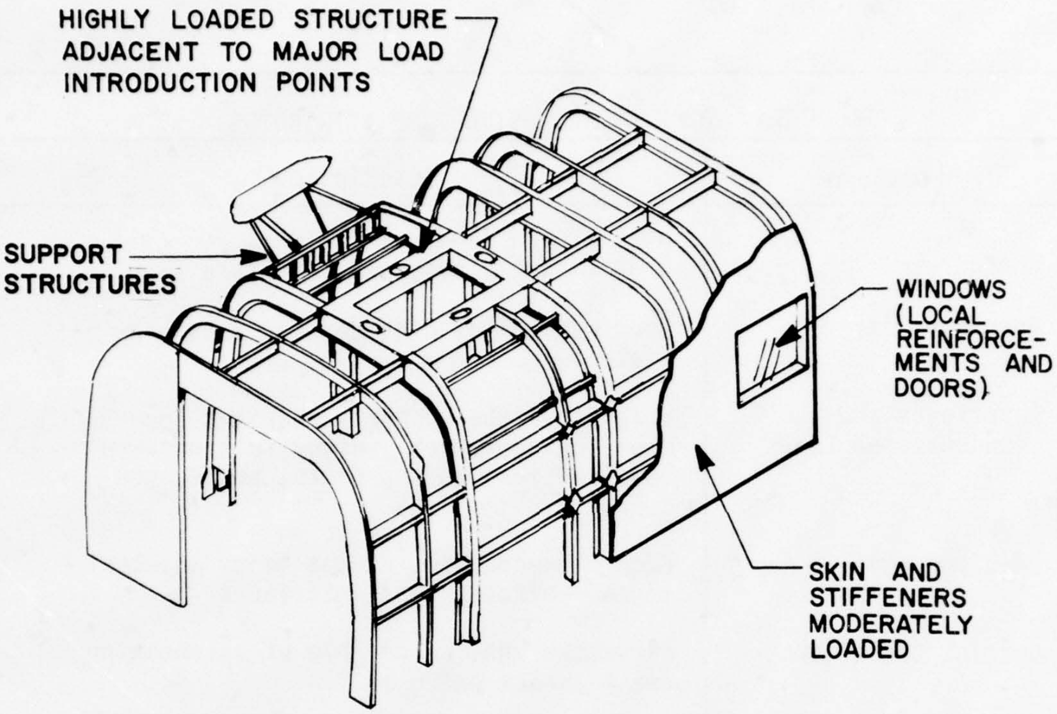


Figure 15. Upper Fuselage Structure

TABLE 7. UPPER FUSELAGE STRUCTURE ATTRIBUTES

Attribute	Description
Contour	Constant cross section or mild compound curvature
Accessibility	Excellent, particularly in passenger compartment
Interfaces and Concentrated Loads	Very heavy structure adjacent to major load introduction points. Bracketry-type supports required for seat and equipment installations
Load Intensities	Moderate with high load intensities at major load introduction points
Special Constraints	Primary structure members highly loaded

and engine support structures. In conventional metal structures these members are normally machined forgings. Heavy structural members are also required adjacent to door and window cutouts. Composite structures concepts are described in Tables 8 and 9.

Lower Fuselage Structure

Lower fuselage structures are of moderately loaded semi-monocoque construction with beam and/or bulkheads spaced to support the floor (Figure 16 and Table 10). A variety of interfaces are common, ranging from lightly loaded equipment supports and moderately loaded cargo tie-downs to heavily loaded cargo sling and landing gear supports. The lower fuselage may also contain fuel tanks, and in such cases, a flush interior surface is required. Watertight construction is a design requirement for amphibious helicopters. A certain degree of crushability is also required to attenuate vertical crash impacts. Composite structures concepts are described in Table 11.

Rear Fuselage

The rear fuselage encompasses the transition area between the cabin and the tail cone (Figure 17 and Table 12). It is designed to support empennage and tail landing gear loads, and as a result of its large cross-sectional area, tends to be of minimum gage construction. The rear fuselage may also contain fuel tanks, and in such cases, tank supports, bulkheads and partitions are required. Composite design concepts are described in Table 13.

Tail Cone

Tail cones are exemplified by structural simplicity (Figure 18 and Table 14). They are essentially tapered cylinders possessing circular or oval cross sections. Depending on the size and load intensities, the structure may be either semi-monocoque or pure monocoque. Interfaces are primarily limited to splice fittings, tail rotor drive shafts and provisions for housing miscellaneous equipment. Composite design concepts are described in Table 15.

Tail Rotor Pylon

The tail rotor pylon is a box-beam type structure that supports the tail rotor, and depending on aircraft configuration, the horizontal stabilizer, intermediate gearbox and tail skid (Figure 19 and Table 16). The cross section is a teardrop with light fairings comprising the leading and trailing edge sections. At the junction to the tail cone, abrupt changes in contour frequently result in fittings with complex geometry.

Equipment installations in the tail rotor pylon are minimal and limited to electrical antennae and flight controls. Composite design concepts are described in Table 17.

TABLE 8. STRUCTURAL DESIGN CONCEPTS - UPPER FUSELAGE

Sub-Component	Composite Concept I	Composite Concept II	Metal Baseline
Skin	Monolithic; woven Kevlar/epoxy	Sandwich; woven Kevlar/epoxy facings with Nomex honeycomb core	Aluminum sheet
Stiffeners and Stringers	Closed section; cross-plied graphite/epoxy hat-shaped members with polyurethane foam covers	None	Open section aluminum channel members
Frames and Beams	Closed section; cross-plied graphite/epoxy hat members with polyurethane foam cores	Sandwich; cross-plied graphite/epoxy facings with Nomex honeycomb core	Open section built-up channels and I-beams fabricated from aluminum sheet and extrusions
Fittings	Cross-plied graphite epoxy	Machined aluminum forgings	Machined aluminum forgings
Assembly Method	Co-cured	Bonded	Riveted
Source	Reference 2	Reference 1	Sikorsky UH-60A

TABLE 9. STRUCTURAL DESIGN CONCEPTS - TRANSMISSION SUPPORT

Sub-Component	Composite Concept I	Composite Concept II	Metal Baseline
Skin	Monolithic; woven Kevlar/epoxy	Sandwich; woven Kevlar/epoxy facings, Nomex core	Aluminum sheet
Stiffeners and Stringers	Local honeycomb areas and graphite/epoxy tape strip reinforcement	None	Open section aluminum channels
Frames and Beams	Open section I-beams; monolithic graphite/epoxy	Sandwich; graphite/epoxy facing, Nomex core	Open section aluminum I-beams built-up from sheet and extrusions
Fittings	Graphite/epoxy reinforcement within basic frames and beams	Machined aluminum forgings	Machined aluminum forgings
Assembly Method	Bonded	Bonded and bolted	Riveted
Source	Reference 3	Reference 1	Sikorsky UH-60A

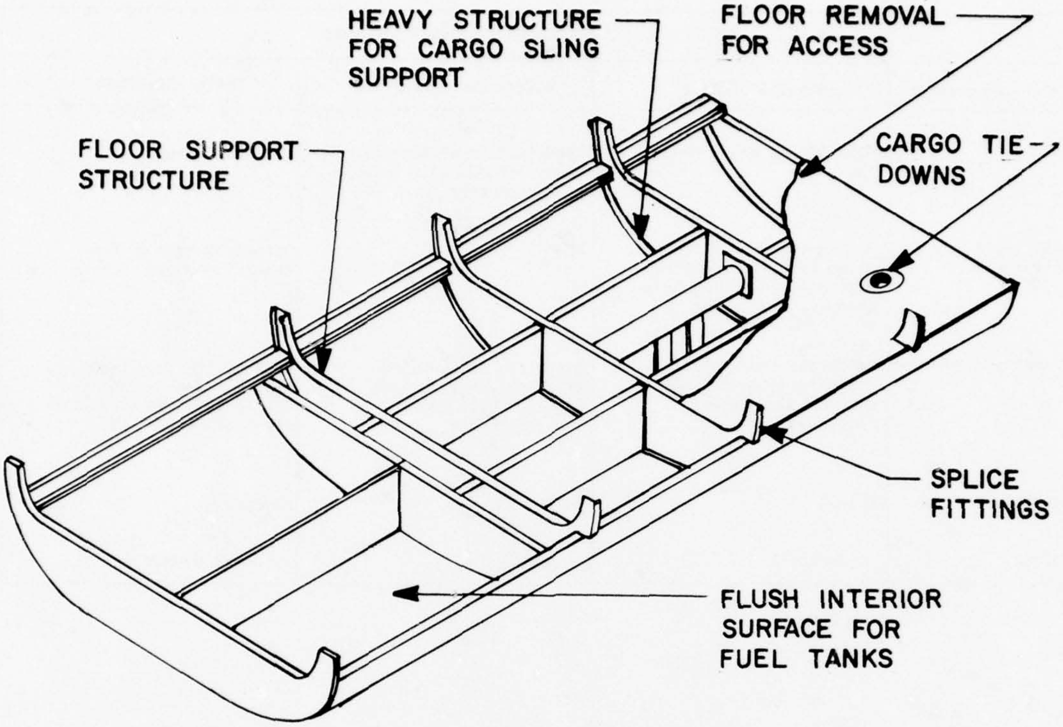


Figure 16. Lower Fuselage Structure

TABLE 10. LOWER FUSELAGE STRUCTURE ATTRIBUTES

Attribute	Description
Contour	Constant cross section or mild compound curvature
Accessibility	Good after removal of floors. Poor in areas surrounding fuel tanks
Interfaces and Concentrated Loads	Heavy structure adjacent to major load introduction points. Bracketry-type supports required for equipment installations
Load Intensity	Moderate with high load intensities at major load introduction points
Special Constraints	Energy absorption capability needed for crashworthiness

TABLE 11. STRUCTURAL DESIGN CONCEPTS - LOWER FUSELAGE

Sub-Component	Composite Concept I	Composite Concept II	Metal Baseline
Skin	Monolithic; woven Kevlar/ epoxy	Sandwich; woven Kevlar/ epoxy facings with Nomex honeycomb core	Aluminum sheet
Stiffeners and Stringers	Closed section; cross-plied graphite/epoxy hat-shaped members with polyurethane foam cores	None	Open section aluminum channel members
Frames and Beams	Sandwich; cross-plied graphite/epoxy facings and caps with Nomex honeycomb core	Sandwich; cross-plied graphite/epoxy facings and cups with Nomex honeycomb core	Open section built-up channels and I-beams fabricated from aluminum sheet and extrusions
Assembly Method	Bonded	Bonded	Riveted
Source	Reference 2	Reference 1	Sikorsky UH-60A

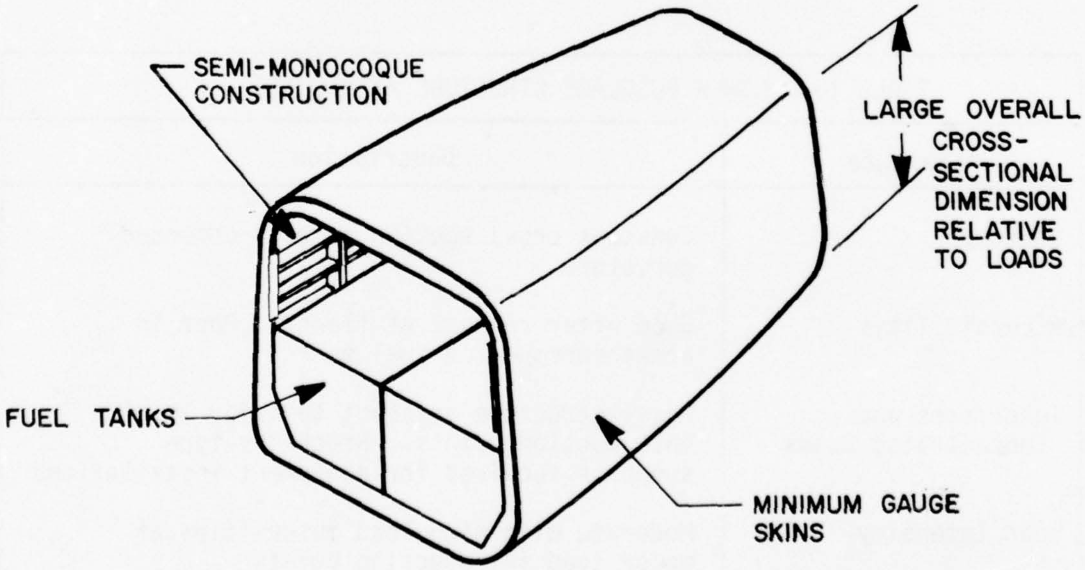


Figure 17. Rear Fuselage

TABLE 12. REAR FUSELAGE STRUCTURE ATTRIBUTES

Attribute	Description
Contour	Mild compound curvature
Accessibility	Excellent, except poor in areas surrounding fuel tanks
Load Intensity	Light, minimum gauge construction
Interfaces and Concentrated Loads	Relatively few interfaces

TABLE 13. STRUCTURAL DESIGN CONCEPTS - REAR FUSELAGE

Sub-Component	Composite Concept I	Composite Concept II	Metal Baseline
Skin	Monolithic; Kevlar/epoxy	Sandwich; Kevlar/epoxy facings, Nomex core	Aluminum sheet
Stiffeners and Stringers	Closed section; Kevlar/epoxy, graphite/epoxy reinforcement	None	Open section; aluminum channels
Frames and Beams	Closed section; Kevlar/epoxy, graphite/epoxy reinforcement	Closed section; graphite/epoxy hat section	Open section; aluminum channels
Bulkheads	Closed section; Kevlar/epoxy, graphite/epoxy reinforcement	Sandwich; Kevlar/epoxy facings with Nomex core	Sandwich; fiberglass/epoxy facings, aluminum honeycomb core
Assembly Method	Bonded	Bonded	Riveted
Source	Synthesized	Reference 1	Sikorsky UH-60A

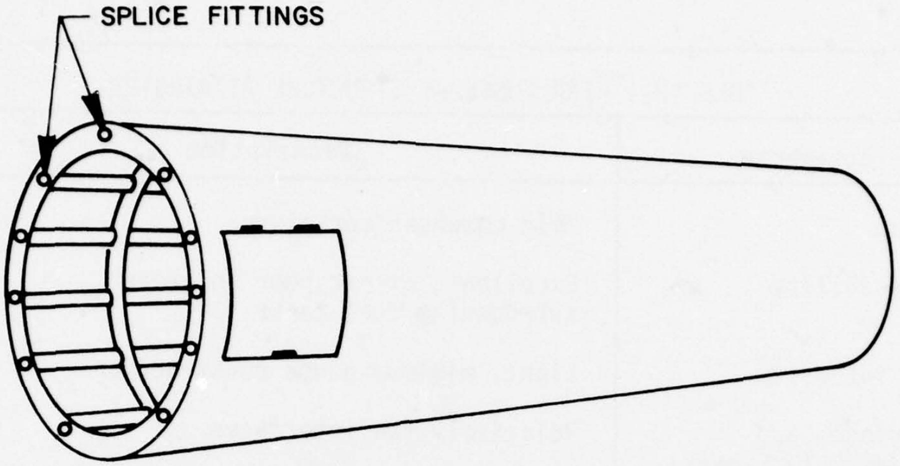


Figure 18. Tail Cone

TABLE 14. TAIL CONE STRUCTURE ATTRIBUTES	
Attribute	**Description**
Contour	Single curvature, wrappable
Accessibility	Generally good, although small size may create restriction
Load Intensities	Moderate to light
Interfaces and Concentrated Loads	Concentrate loads at structural splice points. Relatively few other interfaces.

TABLE 15. STRUCTURAL DESIGN CONCEPTS - TAIL CONE

Sub-Component	Composite Concept I	Composite Concept II	Metal Baseline
Skin	Sandwich; woven Kevlar/epoxy facings with Nomex honeycomb core	Sandwich, cross-plied graphite/epoxy facings with Nomex honeycomb core	Aluminum sheet
Stringers	None	None	Open section aluminum channel members
Frames	Closed section; cross-plied graphite/epoxy hat sections	Open section; cross-plied graphite/epoxy channel sections	Open section aluminum channel members
Assembly Method	Bonded	Bonded	Riveted
Source	Reference 2	Reference 1	Sikorsky UH-60A

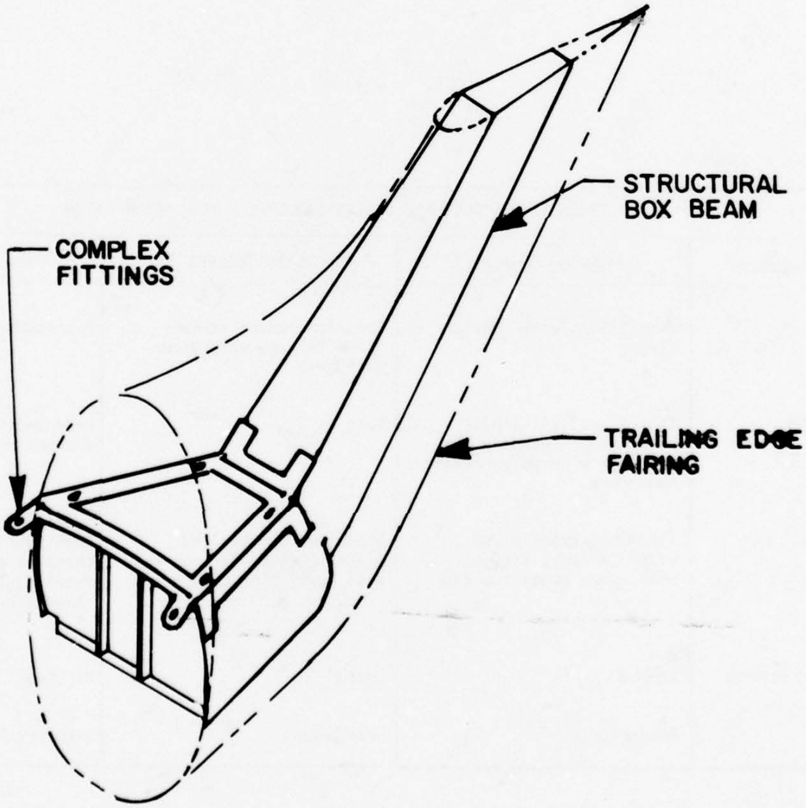


Figure 19. Tail Rotor Pylon

TABLE 16. TAIL ROTOR PYLON STRUCTURE ATTRIBUTES

Attribute	Description
Contour	Mild, except at discontinuities in contour
Accessibility	Small cross section restricts access to interior
Load Intensity	Moderate
Interfaces and Concentrated Loads	Concentrated loads from transmission or stabilizer attachments Minimum number of other interfaces

TABLE 17. STRUCTURAL DESIGN CONCEPTS - TAIL ROTOR PYLON

Sub-Component	Composite Concept I	Composite Concept II	Metal Baseline
Skin	Monolithic, woven Kevlar/epoxy	Sandwich; woven Kevlar/epoxy facings with Nomex honeycomb	Aluminum sheet
Stiffeners	Closed section; cross-plied graphite/epoxy hat sections with polyurethane foam core	None	Open section aluminum channel members
Bulkheads and Spars	Sandwich; cross-plied graphite/epoxy facings with Nomex honeycomb core	Sandwich; cross-plied graphite/epoxy facings with Nomex honeycomb core	Open section built-up channels and I-beams fabricated from aluminum sheet and extrusions
Assembly Method	Bonded	Bonded	Riveted
Source	Reference 2	Reference 1	Sikorsky UH-60A

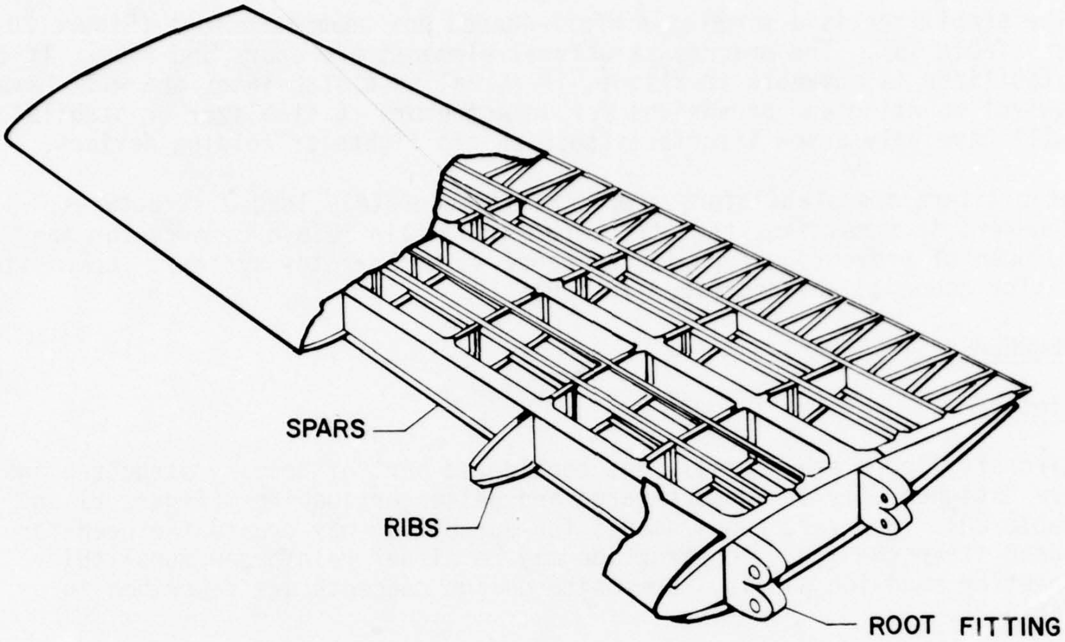


Figure 20. Stabilizer

TABLE 18. STABILIZER/STABILATOR STRUCTURE ATTRIBUTES	
Attribute	**Description**
Contour	Mild, wrappable
Accessibility	Enclosed construction restricts access to interior
Load Intensity	Light to moderate
Interfaces and Concentrated Loads	Concentrated loads at mounting points. Very few interfaces.
Other Constraints	Structures may be sensitive to dynamic resonances.

Stabilizer/Stabilator

The stabilizer is a simple, airfoil-shaped box beam structure (Figure 20 and Table 18). The primary structural elements are spars and ribs. If the stabilizer is moveable in flight, it is called a stabilator and will have a swivel mounting and provisions for an actuator. A stablizer or stabilator will have only a few interfaces such as tip lights or folding devices.

Stabilizers and stabilators are light to moderately loaded structures. However, in some cases the structure may contain reinforcements for the purpose of preventing dynamic resonance with the rotor systems. Composite design concepts are described in Table 19.

SECONDARY STRUCTURE

Floors

Aircraft floors are normally not considered part of primary structure and are designed only to support cargo and passenger loadings (Figure 21 and Table 20). However, requirements for durability may create the need for added strengthening. Construction may be either reinforced monolithic sheet or sandwich panels. Composite design concepts are described in Table 21.

Fairings and Light Cowlings

Fairings and light cowlings are nonstructural and designed primarily to meet aerodynamic pressure and handling loads (Figure 22 and Table 22). (Major cowling, such as that enclosing the engines and transmission and containing large access doors and possibly work platforms, is treated under the category of doors.) Construction may be monolithic, stiffened sheet or sandwich. Minimum gage materials are common because of the light loading. Attachment to major structure may be via hinges or fasteners. Composite design concepts are described in Table 23.

Doors and Major Cowling

Aircraft doors encompass a wide variety of configurations, each having unique attributes and design requirements. Typical types of doors, some of which are illustrated in Figures 23, 24 and 25, are:

Crew and Personnel Door

Air Stair

Sliding Door

Cargo Ramp

Equipment Compartment Door

Access Door and Work Platform

TABLE 19. STRUCTURAL DESIGN CONCEPTS - STABILIZER/STABILATOR

Sub-Component	Composite Concept I	Composite Concept II	Metal Baseline
Skin	Sandwich; woven Kevlar/epoxy facings with Nomex honeycomb core	Sandwich; cross-plied graphite/epoxy with Nomex honeycomb core	Aluminum sheet
Stiffeners	None	None	Open section; aluminum channel members
Spars and Ribs	Sandwich; cross-plied graphite/epoxy facings with Nomex honeycomb core	Monolithic - cross-plied graphite/epoxy	Open section; built-up channels and I-beams fabricated from aluminum sheet and extrusions
Assembly Method	Bonded	Bonded	Riveted
Source	Reference 2	Reference 1	Sikorsky UH-60A

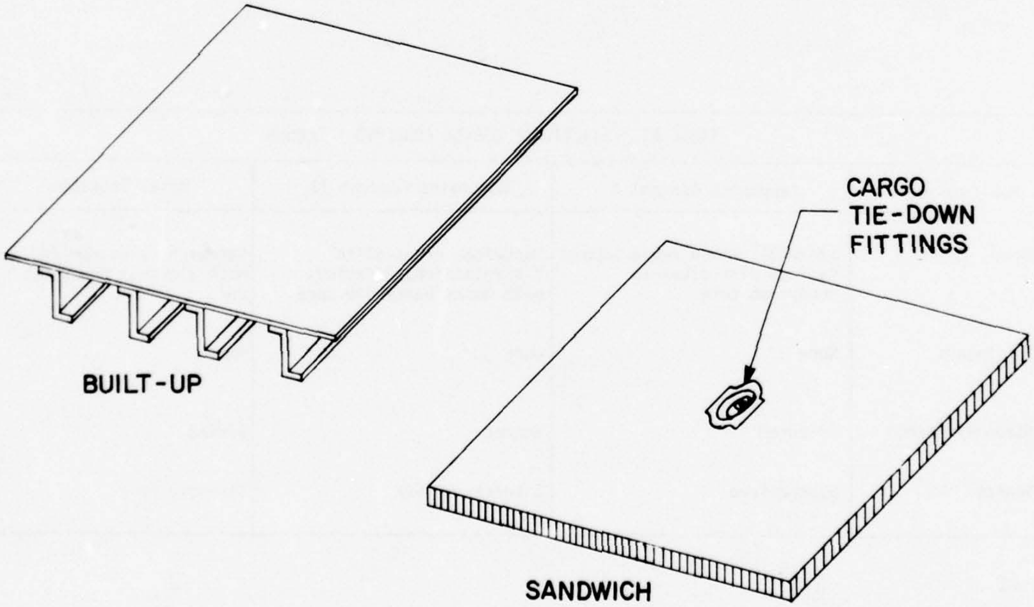


Figure 21. Floors

TABLE 20. FLOOR STRUCTURE ATTRIBUTES

Attribute	Description
Contour	Flat
Load Intensity	Moderate
Accessibility	Excellent when panels are removable
Interfaces and Concentrated Loads	Local reinforcements for cargo tie-down and seat mountings

TABLE 21. STRUCTURAL DESIGN CONCEPTS - FLOORS

Sub-Component	Composite Concept I	Composite Concept II	Metal Baseline
Skin	Sandwich; woven Kevlar/epoxy facings with aluminum honeycomb core	Sandwich; cross-plied fiberglass/epoxy facings with Nomex honeycomb core	Sandwich; aluminum facings with aluminum honeycomb core
Stiffeners	None	None	None
Assembly Method	Co-cured	Bonded	Bonded
Source	Synthesized	Sikorsky UH-60A	Sikorsky SH-3

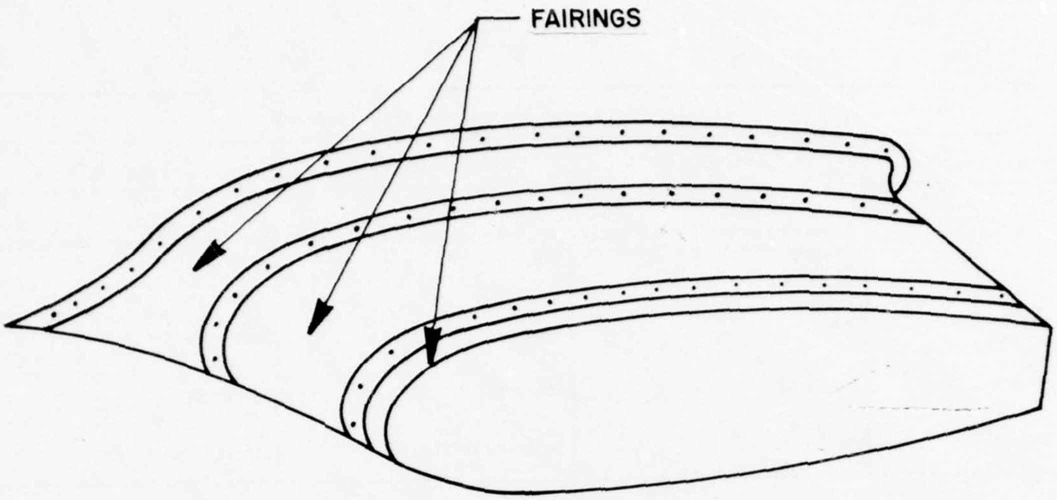


Figure 22. Fairing

TABLE 22. FAIRING STRUCTURE ATTRIBUTES	
Attribute	Description
Contour	Ranges from flat to extreme compound curvature depending on configuration
Accessibility	Usually good
Load Intensity	Light
Interfaces	Attachment points

TABLE 23. STRUCTURAL DESIGN CONCEPTS - FAIRINGS

Sub-Component	Composite Concept I	Composite Concept II	Baseline
Skin	Sandwich; woven Kevlar/epoxy facings with Nomex honeycomb core	Monolithic; woven Kevlar/ epoxy	Monolithic; woven fiberglass/ epoxy
Stiffeners	None	None	None
Assembly	Co-cured	Co-cured	Co-cured
Source	Reference 1	Sikorsky UH-60A	Sikorsky SH-3, etc.

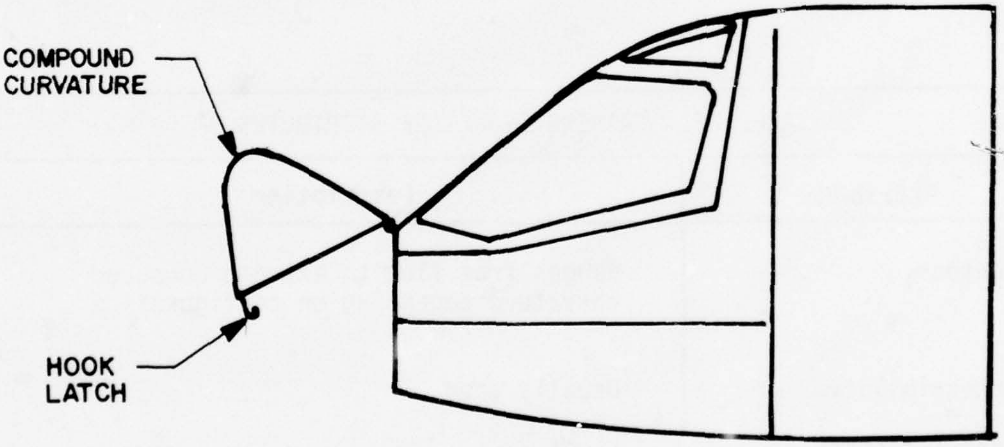


Figure 23. Nose Compartment Door

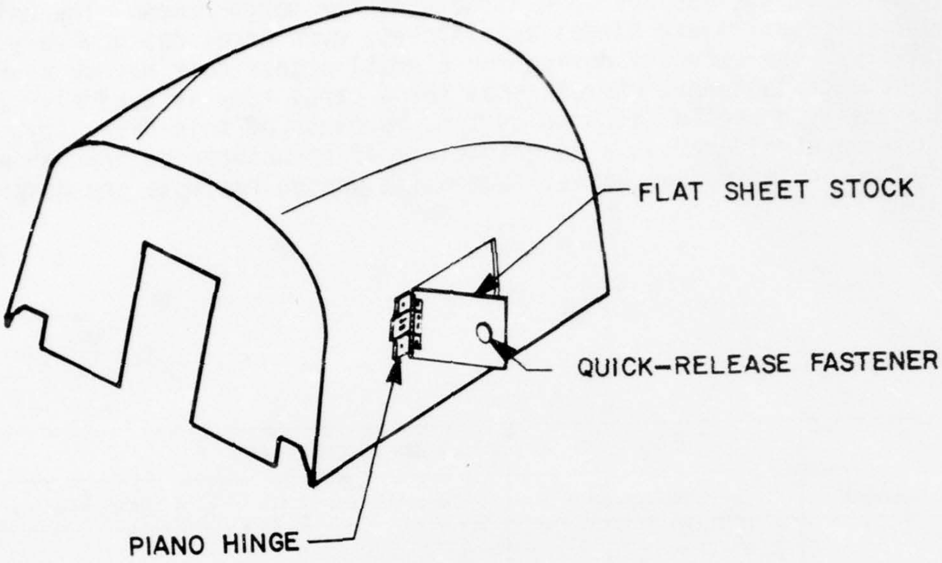


Figure 24. Simple Access Door

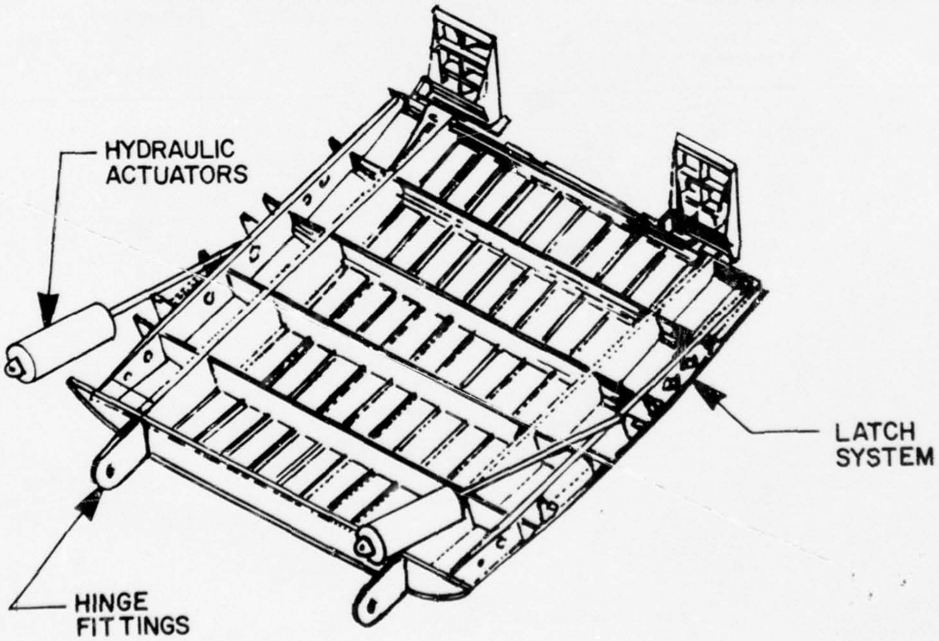


Figure 25. Cargo Ramp

Construction ranges from unreinforced monolithic sheet for small access doors to built-up fuselage type structures for cargo ramps. The only common characteristics are hinges and latches; even here, designs vary substantially. The security device for a small access door may be a single quick-release fastener, whereas that for a cargo ramp is typically a complex, hydraulically operated latching system. Because of this diversity, doors will not be considered as a generic class of structures but rather will be treated on an individual basis. Composite design concepts are described in Table 24.

TABLE 24. STRUCTURAL DESIGN CONCEPTS - DOORS

Sub-Component	Composite Concept I	Composite Concept II	Metal Baseline
Skin	Sandwich; woven Kevlar/epoxy with Nomex honeycomb core	Monolithic; woven Kevlar/epoxy	Aluminum sheet
Stiffeners	None	Closed section; cross-plied Kevlar/epoxy hat sections	Closed section; aluminum hat section
Assembly Method	Co-cured	Bonded	Spot-welded
Source	Reference 1	Synthesized	Sikorsky SH-3

SERVICE EXPERIENCE WITH AIRFRAME STRUCTURES

An investigation was conducted to assess the R&M experience of airframe structures in service. Special attention was given to bonded structures and composites. The investigation included a review of published data on current-inventory Army helicopters, and with reference to bonded structures and composites, an analysis of in-house data on Sikorsky helicopter models, an examination of fixed-wing aircraft experience,and visits to two Army helicopter depots.

ASSESSMENT METHODS

Because of their monolithic design, fuselage structures present more difficult problems of reliability assessment than do aircraft subsystems consisting of highly differentiated components. Attempts to use field service data for the reliability analysis of aircraft structures encounter particular problems.

Very abbreviated descriptions of failures conveyed via the standard reporting systems comprise the bulk of recorded service experience with military aircraft. Nevertheless, for the majority of components on an aircraft, it is possible to obtain a reasonable understanding of the types of failures occurring in service. In addition to the coded descriptions of each failure, records of individual parts replaced in the process of maintenance often provide further insights. Thus, the report of a "leaking" valve, combined with a list of specific seals replaced in the course of repair, provides a good indication of the failure that occurred.

This kind of visibility is lacking with airframe structures, however. In order to assess the nature of structural failures and induced damage, it is important to know not only the general type of defect (crack, dent, puncture, etc.) but also the location of the damage, the structural elements involved and the extent and severity of the fault. A record of a "crack" in the "tail pylon" of a helicopter - the level of detail typically contained in field reports - is quite meaningless from the standpoint of reliability assessment, except to record the occurrence of the event. It is unknown whether the crack occurred in a superficial area such as the skin and was repaired by simple stop drilling, or occurred in a major structural element of the pylon such as the spar and required a complex structural repair. From the standpoint of reliability, the two events are not at all equivalent.

The number of man-hours involved in the maintenance action may provide some clue to the scope of the repair task, and therefore the degree of structural damage, but man-hours may also be a very misleading indicator of task complexity. Furthermore, because most structural repairs are accomplished with common hardware and bulk materials (aluminum sheet, fiberglass cloth, rivets, etc.) there is no record of replaced parts that can be used to assess the nature of the failure or damage. As a consequence, service experience with fuselage structures is among the most poorly documented of all aircraft subsystems.

ARMY HELICOPTER AIRFRAME SERVICE EXPERIENCE

In September 1974 the U.S. Army Aviation Systems Command published the results of an investigation of R&M problems with five subsystems of the UH-1 and CH-47 helicopters (Reference 4). The study was based on documented service experience with the two helicopters. The airframe was one of the five subsystems examined.

For the CH-47 helicopter, the study was based on maintenance data collected by the U.S. Army Aviation Test Board at Fort Rucker. The data covered 4,132 flight-hours accumulated over a 16-month period ending September 1970. For the UH-1 helicopter, data collected on the U.S. Air Force UH-1F encompassing 42,869 flight-hours over a 12-month period ending February 1972 was used. Other records of the U.S. Army and of the respective aircraft contractors were used to augment these two principal data sources.

Unscheduled maintenance data for the airframe systems of the two aircraft disclose remarkable similarities as shown in Table 25. The frequency of unscheduled maintenance is of course greater for the much larger airframe of the CH-47, but the breakdown of maintenance by elements of the airframe is nearly identical. A representative distribution of unscheduled maintenance events based on a composite of the service experience with these two aircraft is shown in Figures 26 and 27. Several conclusions are apparent.

TABLE 25. PERCENTAGE BREAKDOWN OF UNSCHEDULED MAINTENANCE EVENTS FROM REFERENCE 4

	UH-1F	CH-47C
Unscheduled Maintenance Events/ 1,000 Flight-Hours	242.4	536.1
Airframe Percent of Total Aircraft	31.4	37.3
Percent of Airframe		
Secondary Structure	84.7	80.8
Primary Structure	15.3	19.2
Percent of Primary Structure		
Skin	41.6	54.2
Structure	46.1	45.8
Rivets/Hardware Percent of Total	50.0	38.3

---

[4] Barrett, L.D., and Aronson, R. B., RELIABILITY AND MAINTAINABILITY PROGRAM FOR SELECTED SUBSYSTEMS AND COMPONENTS OF CH-47 AND UH-1 HELICOPTERS, Boeing Vertol Company, Report Number D210-10846-1, U. S. Army Aviation Systems Command, St. Louis, MO, September 1974.

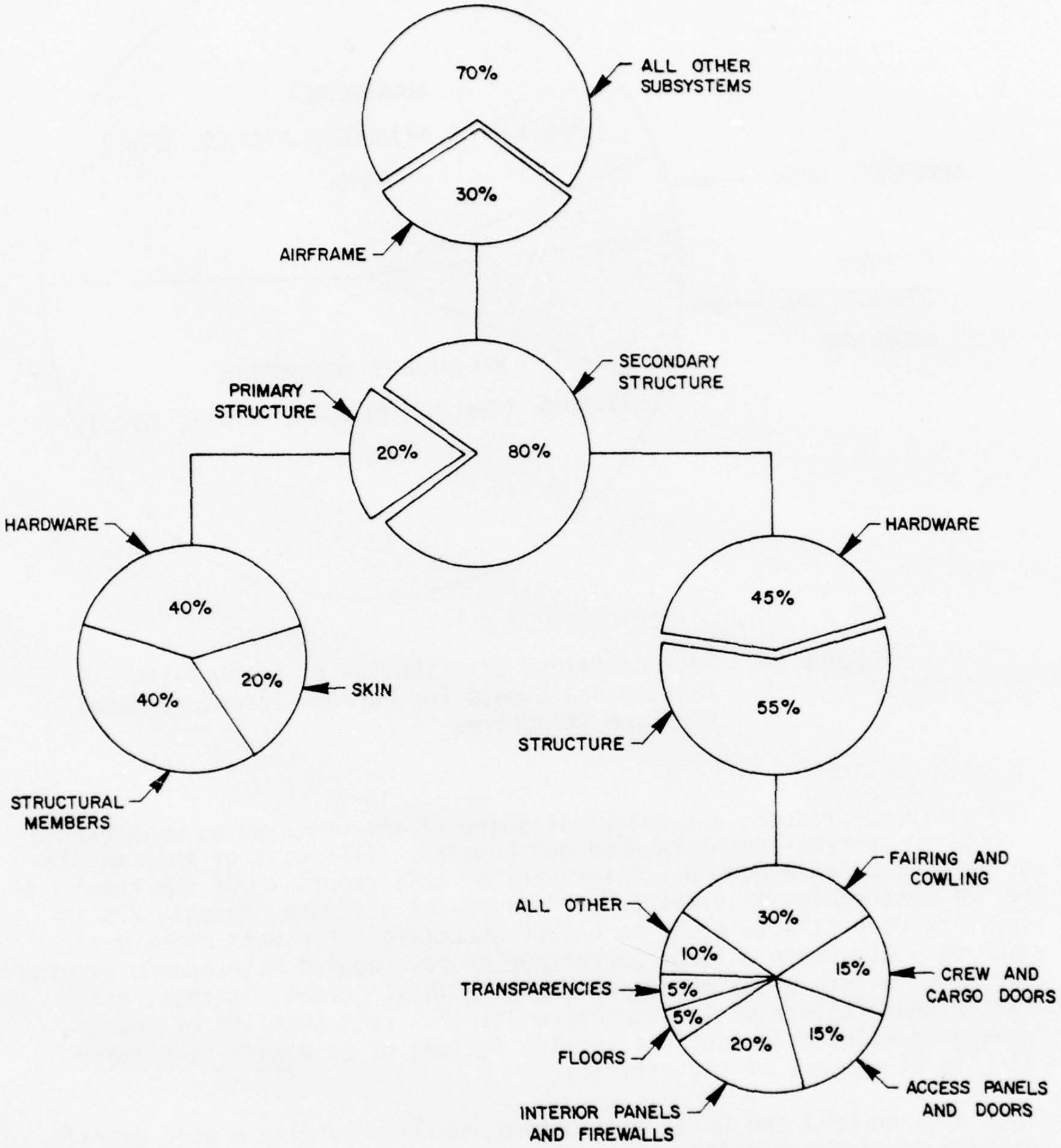


Figure 26. Representative Distribution of Unscheduled Maintenance Events for Current-Inventory Helicopters

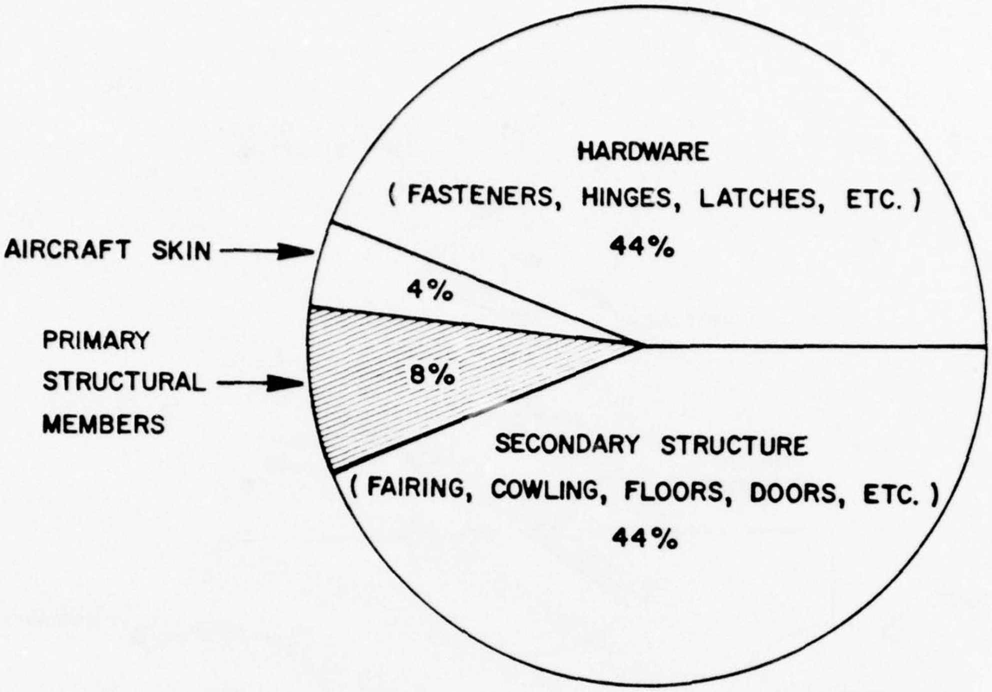


Figure 27. Representative Distribution of Unscheduled Maintenance Events for Current-Inventory Metal Airframe Structures

The airframe produces a substantial share of the unscheduled maintenance events on current-inventory Army helicopters. (The cost of this maintenance will be examined in a later part of this report.) Of the total number of unscheduled maintenance actions on the airframe, roughly 20% involve primary structure,80% secondary structure. For both primary and secondary structure a large percentage of unscheduled maintenance events involves failure of attaching hardware (rivets, screws, latches, etc.) rather than failure of the structure itself. Less than 10% of the unscheduled maintenance actions involve failure of or damage to primary structural members of the airframe.

This data on UH-1 and CH-47 helicopters, while providing a good overall picture of airframe maintenance on Army aircraft, suffers some of the shortcomings of field data referred to earlier. Within the limitations of the code systems used to record aircraft maintenance in the field, it does identify the general type of structure and the modes of damage or failure that each action involves. It does not identify specific components of the airframe, the location and nature of the damage or failure, or the types of repairs made, however.

Environmental Effects

An effort was made to establish the effects of various environments on the reliability of helicopter airframe structures. It was learned early in the program that a comprehensive investigation of the effects of environment on Army aircraft was being conducted by the Los Angeles Division of Rockwell International for the Army's Applied Technology Laboratory at Fort Eustis. From their investigation, involving extensive surveys of published data as well as independent research, Rockwell had compiled environmental effects information on all the major subsystems of helicopters. The Applied Technology Laboratory provided to Sikorsky computer printouts of this data for the airframe subsystem. Table 26 summarizes the data and ranks the environmental hazards by frequency of inflicted damage. Appendix A contains a detailed breakdown by areas of the helicopter.

TABLE 26. SUMMARY OF ENVIRONMENTAL EFFECTS ON HELICOPTER AIRFRAME STRUCTURES

	Events Per 10^5 Flight-Hours				
	Surface Damage	Structural			Total
Environmental Hazard		Damage	Deformation	Deterioration	
Fluctuating Loads	906	26,865			27,771
Maintenance/Handling	1,171	7,608	149		8,928
Vibration	54	2,052			2,106
Rotor Downwash		1,240	93		1,333
Mechanical Shock		762			762
Moisture/Precipitation		583			583
Aircraft Fluids		583			583
Cleaning Fluids				150	150
Rotor Circulated Sand & Dust		116			116
Crew Damage		86	25		111
Foreign Object Damage		26			26
Total	2,131	39,921	267	150	42,469

The four categories of damage were derived by combining individually reported failure modes as follows:

Surface Damage

- Scratched
- Crazed
- Worn
- Nicked
- Pitted
- Scored

Structural Damage

- Broken
- Cracked
- Dented
- Punctured
- Separated
- Torn
- Sheared

Structural Deformation

- Collapsed
- Buckled
- Distorted
- Warped
- Bent

Structural Deterioration

- Deteriorated
- Overheated
- Corroded

As shown, the vast majority of all damage events is caused by just a few environments. It should be stressed that the data came from a limited number of sources, many of which focused on a particular subject area or problem, and was considered by Rockwell to be neither complete nor necessarily representative of the true reliability of the components it covers. Moreover, the sources used frequently failed to report the cause of failure, and it was necessary for Rockwell in many cases to judge whether a failure was environmentally caused and to establish the environment involved.

Nearly two-thirds of the damage events reported were said to have occurred as a result of fluctuating loads. The specific nature of these events could not be determined from the data, but most of them probably involved minor failures such as popped rivets and fatigue cracks. This assumption seems reasonable for the riveted metal construction typical of helicopters now in service. Maintenance and handling, and vibration are the two other environments showing a significant effect on damage rate. Again, however, the specific nature of these events could not be determined from the data.

BONDED STRUCTURES AND COMPOSITES

Military aircraft provide the largest experience base for bonded structures and composites. Most of the present airframe structure consists of bonded aluminum honeycomb panels and fiberglass components. The use of advanced composites for primary structure is very limited and confined almost entirely to control surfaces on high-performance fixed-wing aircraft. None of the helicopters in service with the U.S. Military employ advanced composites in significant quantity.

A survey was made of both fixed-wing and helicopter experience with composites. For fixed-wing aircraft, service data was extracted from a study of advanced composite structures conducted by the Northrop Corporation for the Air Force Flight Dynamics Laboratory. Helicopter experience with composites was assessed from an analysis of R&M data on Sikorsky aircraft in service and from surveys of U.S. Army depots that are overhauling and repairing fleet aircraft. Other published data was examined, as reported in the Bibliography and List of References, but nothing of significance to this study was found.

Fixed-Wing Aircraft Experience

The difficulty of assessing the in-service reliability of aircraft structures confronted the Northrop Corporation in a study of advanced composite structures for the U.S. Air Force Flight Dynamics Laboratory. In one of the most comprehensive investigations of the subject to date, Northrop surveyed the service experience with fuselage structures on a variety of Air Force aircraft, including the F-111, F-104, A-37, A-7D and Northrop's YF-17 prototype. The survey focused on service experience with both composites and conventional metal structures and encompassed extensive searches of the Air Force AFM 66-1 Data System, an analysis of Northrop's own data on the YF-17, and direct interviews with Air Force and NASA personnel.

In their fourth quarterly report on the contract, Reference 5, Northrop observed that "Although considerable information is available through the AFM 66-1 system, the data do not indicate the severity of the damage nor the specific cause of the damage." This deficiency was overcome by using the well-documented history of the YF-17 flight test program and engineering data obtained from interviews with Air Force and NASA personnel to identify the major sources of ground handling damage and the relative susceptibility of various structures to this type of damage. Meaningful quantitative measures of the frequency and severity of the various damage modes could not be developed, however.

---

[5] Labor, J.D., SERVICE/MAINTAINABILITY OF ADVANCED COMPOSITE STRUCTURES, Quarterly Progress Report Number 4, Northrop Corporation Report Number 77-157, Contract F33615-76-C-3142, U.S. Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio, November 1977.

Table 27 summarizes the conclusions of the fixed-wing study relative to the types of damage that composite structures are expected to receive from ground handling and maintenance. In general, four sources of damage were found to be significant:

1. Surface impact
2. Edge and corner impact
3. Foot traffic
4. Fastener damage

Survey of Sikorsky Experience

Composite structures of both monolithic and honeycomb sandwich design are used on all of Sikorsky's operational aircraft and more extensively on several new models in development and early production during the period of this program. The types of components range from simple fiberglass fairings to such items of primary structure as the cockpit canopy for the CH-53 helicopter and the all-composite stabilator for Sikorsky's new commercial helicopter, the S-76. Composites are also used extensively by Sikorsky in the construction of main and tail rotor blades for helicopters.

Composites in the airframe structures of Sikorsky models operational prior to 1979 were confined almost exclusively to fiberglass, both in monolithic form and in the facings of sandwich panels. Sandwich core was either aluminum or Nomex honeycomb. Fiberglass was also used over structural foam in the construction of formers and stiffeners.

In addition to fiberglass, Kevlar was being introduced at the time of this program in the airframe structures for Sikorsky's newest aircraft: the Army UH-60A Black Hawk, Navy SH-60B, Navy/Marine Corps CH-53E and Commercial S-76. The types of construction with Kevlar are basically similar to those with fiberglass: flat or contoured skin panels, stiffened panels and honeycomb sandwich structures. Graphite and boron, the advanced composites, have found limited applications to date. In the airframe, boron/epoxy has been used to stiffen the cockpit support beams for the UH-60A and to reinforce the tail cone for the CH-54. Outside of its use in the construction of rotor blades, graphite/epoxy has been used in one limited application: the stabilator for the S-76. At the time of this program, a wide range of R&D programs at Sikorsky were exploring further uses of advanced composites, and applications to current aircraft, such as the rear fuselage section of the UH-60A, were nearing production development.

Service experience with composite structures on Sikorsky helicopter models was investigated. The service histories of the Navy SH-3D, Marine Corps CH-53D and Army UH-60A were examined. Data for this study was obtained from the Navy's 3-M System for the SH-3D and CH-53D helicopters and from the Army's RAM/LOG System for the UH-60A helicopter.

TABLE 27. SUMMARY OF COMPONENT DAMAGE SUSCEPTIBILITY FROM FIXED-WING AIRCRAFT STUDY

Damage Mode	Source/Cause	Frequency and/or Severity	Effect of Configuration	Effect of Location	Effect of Orientation
Surface Damage (Nicks, Punctures, Dents, etc.)	. Dropped Tools . Dropped Equipment . Work Stands	Dropped tools most significant source of damage.	Thin-faced panels suffer most damage. Stiff panels damaged at lower energy levels than flexible ones.	Damage most prevalent in high traffic areas during refueling, rearming, etc. Components high on aircraft least affected. Extremities (wing tips, etc.) most susceptible to ground impact.	Top surfaces and horizontal surfaces most vulnerable. Bottom and vertical surfaces see much less damage.
Heel Damage (Dents, etc.)	. Foot Traffic	Frequent in no-step areas adjacent to designated walkways.	Same as above.	Areas adjacent to designated walkways.	Top, horizontal surfaces primarily.
Disbonding of Face Sheets from Core Material	. Foot Traffic . Impact	Not definitely established as significant damage mode.			
Disbonding of Stiffeners	. Impact	Can occur at low energy levels with no visible surface damage.			
Edge and Corner Damage	. Dropped Component . Impact by Work Stand, Hoist Cable, etc.	Dropped components most significant source of damage.	Removable components most vulnerable, especially if heavy and awkward to handle.	Components high on aircraft and in areas where handling is awkward are most vulnerable.	
Fastener Damage (Hole Elongation, Delamination, etc.)	. Fastener Misalignment, Over-Torque or Pull-Thru		Damage resistance improves for thicker panels and panels using edge doublers.	Locations awkward to reach receive more damage. Wear proportional to frequency of removal.	Vertical panels suffer more damage because of greater tendency for misalignment during installation.
Major Structural Damage	. Impact by Ground Vehicles			Structure on lower part of aircraft; protuberances most vulnerable.	

The calendar periods and number of flight-hours coverd by the respective data samples are listed below:

Service History Data Base

Model	Calendar Period	Flight-Hours
SH-3D	April 1971 - June 1974	74,649
CH-53D	January 1975 - December 1976	25,829
UH-60A	November 1974 - December 1977	1,889

All of the bonded panels and composite structures on these three aircraft were identified and cross-referenced to the code systems by which the field data is stored: work unit codes for 3-M and math model codes for RAM/LOG. Computer printouts of each file were reviewed, and the R&M statistics recorded for each component were extracted and tabulated. The data on each component was then screened to eliminate obvious errors and to reduce the data to those events that reflected the occurrence of structural failure or damage. Thus, reports of removal to facilitate maintenance, or of discrepancies such as lack of lubrication, were disregarded.

Within the limitations of the failure codes used for reporting field maintenance, it was possible to identify five basic types of damage of a structural nature. In addition to the coded information, the data on the UH-60A provided narrative descriptions of failures and maintenance, but it was also the smallest of the three samples used and therefore contained a very small number of reports of interest to the study.

Types of damage revealed by the service data and considered pertinent to the study are listed at the headings of the five right-most columns of Table 28. The table is organized by types of construction and aircraft location. For example, all components of monolithic fiberglass construction will be found grouped and listed by general areas of the aircraft. This was done in an attempt to reveal patterns of damage or failure related to aircraft location.

The tabulated data is quite sketchy, especially in the case of the UH-60A which had at the time accumulated fewer than 2,000 hours of test flying. Few significant patterns are evident. The damage modes most frequently reported are bent or broken, cracked, and loose or missing hardware. Corrosion is reported fairly consistently, indicative of the metal fasteners and hinges which are common among these components. Structures incorporating metal frames appear to have a slightly higher incidence of corrosion than do those incorporating composite frames, as do sandwich structures employing aluminum versus Nomex honeycomb.

With respect to location, components installed in the engine and transmission nacelle areas and the aft fuselage appear to suffer the highest damage rates. The level of maintenance activity in the vicinity of major dynamic components such as the engines, transmissions and rotors may account for this.

TABLE 28. SUMMARY OF SERVICE EXPERIENCE WITH BONDED STRUCTURES AND COMPOSITES ON SIKORSKY HELICOPTER MODELS

Aircraft Location	Aircraft Component/Structure	Code	Aircraft Model	Damage Rate/10^5 Flight-Hours				
				Bent/ Broken	Cracked	Nicked/ Chipped	Corroded	Loose/ Missing Hardware
Monolithic Fiberglass Construction								
Forward Fuselage/ Cockpit Canopy	Covers	1122F	SH-3D	28	127	1	29	43
	Housing, Hover Light	11216	SH-3D	1	1		8	
	Reservoir, Windshield Washer	12315	CH-53D	8				4
Center Fuselage	Liner, Fuel Tank	1131B	SH-3D	20	40	1	15	21
	Sponson, Leading Edge	11213	CH-53D	27	19		31	4
	Sponson, Trailing Edge	11222	CH-53D	35	27		15	
	Fairing, Aux Fuel Tank Support	46712	CH-53D	4	4		12	39
Cabin	Duct, Air	41161	SH-3D	1				4
	Panel, Drip	11141	CH-53D	120	35			43
	Duct, Air	12151	CH-53D	27	35			
Aft Fuselage	Fairing, Stabilizer	11513	SH-3D	5	21	1	32	11
	Fairing, Aft Ramp Opening	11168	CH-53D	46	46		8	
Tail Pylon	Cover, Hinge	11318	CH-53D	39	39		4	4
	Fairings, Tail Rotor Gearbox and Stabilizer	11194	CH-53D	101	74		35	163

TABLE 28 (Continued)

Aircraft Location	Aircraft Component/Structure	Code	Aircraft Model	Damage Rate/10^5 Flight-Hours				
				Bent/ Broken	Cracked	Nicked/ Chipped	Corroded	Loose/ Missing Hardware
Monolithic Kevlar Construction								
Cockpit	Shield Glare	10A50	UH-60A					
Center Fuselage	Fairing, Step, Main Landing Gear	3A2A8	UH-60A		53			
	Fairing, Oleo	3A2A9	UH-60A					
Cabin	Duct, Air	11A3A8	UH-60A		159			
Fiberglass Skins with Metal Frames								
Engine and Transmission Nacelles	Fairing, Engine & Transmission	1123Q	SH-3D	72	419	4	52	214
	Duct, Air Intake, Engine	29211	SH-3D	15	113			13
Main Rotor Pylon	Fairing, Main Rotor Pylon	1131J	CH-53D	15	27		19	15
	Housing, Pylon	11315	CH-53D	85	147		19	77
Kevlar Skins with Metal Frames								
Main Rotor Pylon	Fairing, Fixed	3A4A10	UH-60A		53			

TABLE 28 (Continued)

Aircraft Location	Aircraft Component/Structure	Code	Aircraft Model	Damage Rate/10^5 Flight-Hours				
				Bent/ Broken	Cracked	Nicked/ Chipped	Corroded	Loose/ Missing Hardware
Fiberglass Skins with Fiberglass Formers and/or Fiberglass Covered Structural Foam Core Stiffeners								
Forward Fuselage/ Cockpit Canopy	Cockpit & Canopy Structure	11110	CH-53D	29	17		57	23
	Door, Battery Access	11111	SH-3D		16		54	
	Frame, Cockpit Escape Window	1111320	CH-53D	11	6		11	6
Lower Fuselage	Covers	1131B	SH-3D		41		15	19
Rear Fuselage	Cover	1151E	SH-3D	54	312	19	111	142
Engine Inlet	Fairing, Nose, EAPS	2971430	CH-53D		19		4	0
	Door, Intake, EAPS	2971440	CH-53D	8	8			4
	Panel, Forward, EAPS	29714D0	CH-53D					
Engine Nacelle	Panel, Outboard, Forward	2921M	CH-53D	155	143	0	4	19
	Panel, Outboard, Aft	2931Q	CH-53D					
Engine and Transmission Nacelles	Fairing, Panel, Access	1123S	SH-3D	48	99		33	107
	Fairing, Door, Access	1123T	SH-3D	52	33		20	87
Main Rotor Pylon	Housing, Pylon	1131L	CH-53D	43	27		43	27

TABLE 28 (Concluded)

Aircraft Location	Aircraft Component/Structure	Code	Aircraft Model	Damage Rate/10^5 Flight-Hours				
				Bent/ Broken	Cracked	Nicked/ Chipped	Corroded	Loose/ Missing Hardware
Kevlar Skins with Kevlar Formers and/or Kevlar Covered Structural Foam Core Stiffeners								
Main Rotor Pylon	Fairing, Pylon	3A3A4	UH-60A		53			
	Fairing, Aft	3A4A11	UH-60A					
Sandwich Construction (Fiberglass Skins with Nomex or Aluminum Honeycomb)								
Cockpit Canopy	Door, Electronics	11114	CH-53D	17	23	6	177	6
Forward Cabin	Door, Upper	11132	CH-53D	12	19		4	4
Engine Nacelle	Cowling, Detachable	29210	SH-3D	17	50		28	90
Sandwich Construction (Kevlar Skins with Nomex or Aluminum Honeycomb)								
Forward Fuselage	Door	3A1A5	UH-60A	53				
Aft Fuselage	Access Panel, Belly	3A3A1	UH-60A					
Engine Nacelle	Cowling, Engine	2A6	UH-60A					
	Work Platform	2A6A1	UH-60A					
Main Rotor Pylon	Fairing, Main Gearbox	3A4A5	UH-60A					

In attempting to draw meaningful conclusions from this data, one encounters the same problems mentioned in the discussion on fixed-wing aircraft experience. The data is not definitive, and it is impossible to determine the specific modes of damage, their location or their severity. Only general impressions can be gleaned as a result.

Surveys of Army Helicopter Depots

Inquiries were made at AVRADCOM, St. Louis, in an effort to obtain statistical data on in-service experience with composite structures on Army helicopters. Data of this type could not be located, and personnel at AVRADCOM suggested that visits to the Army depots might provide worthwhile information on this subject. Two depot surveys were made.

The first visit was made to the U.S. Army Depot at Corpus Christi, Texas, where UH-1 and AH-1 helicopters are overhauled and repaired. The second visit was made to the depot at New Cumberland, Pennsylvania, which provides this support for the CH-47 and OH-58 helicopters. Each of the two surveys entailed examinations of aircraft structures in various states of damage and repair and detailed discussions with depot personnel. Inquiries focused on the types of damage and field repairs that depot personnel observe on fiberglass components and bonded panels when aircraft are inducted into overhaul. Opinions of depot personnel on the quality of field maintenance were also obtained.

Information gathered from these surveys, while strictly of a qualitative nature, provides a much better impression of service experience with composites than was obtained from the data searches on the fixed-wing aircraft and Sikorsky helicopter models. Equally important, it reflects the experience of Army helicopter operations in the field.

The types of bonded panels and fiberglass components on current-inventory Army helicopters are similar to those on Sikorsky models operational at the time of the surveys. Sandwich panels are primarily aluminum honeycomb with aluminum skins. Titanium skins are used for some of the engine decks and fiberglass skins for one or both faces of some panels. Sandwich panels of Nomex and fiberglass construction are used for some of the cowling and fairings on the H-1 models. Fiberglass components consist of light fairings and covers and a few pieces of large stiffened structure. For each of the four aircraft, Table 29 lists the types of bonded structures and composites that were discussed with depot personnel in the course of the surveys.

The results of the depot surveys are tabulated in Table 30 for the UH-1 and AH-1 helicopters and in Tables 31 and 32 for the CH-47 and OH-58 helicopters respectively. Each table is organized by type of construction and aircraft component. For each component, seven basic damage modes are rated based on the examinations of aircraft and discussions with depot personnel. The reported incidence of damage of each type is rated as heavy, moderate or light using a system of shaded blocks to record the ratings. The absence of shading indicates that the damage mode either is not applicable to a component or was not reported as significant by depot personnel.

TABLE 29. TYPES OF BONDED STRUCTURES AND FIBERGLASS COMPONENTS COVERED BY THE DEPOT SURVEYS

Type of Construction Type of Structure	Aircraft Model			
Type of Component	UH-1	AH-1	CH-47	OH-58
Aluminum Honeycomb Panels				
Primary Structure				
Fuselage Shell				X
Main Beams	X	X		
Frames and Formers			X	
Exterior Fuselage Panels	X	X	X	X
Roof Panels	X	X	X	
Interior Bulkheads	X			
Fuel Cell Compartments/Pods	X	X	X	X
Decks and Floors				
Floor Panels	X	X		X
Engine Decks	X	X		X
Service Decks	X	X		
Walkways			X	
Work Platforms			X	
Secondary Structure				
Interior Compartment Walls/Floors	X	X		
Equipment Bay Shelves	X	X	X	X
Fuel Cell Compartment Liners	X			
Nomex Honeycomb Panels				
Secondary Structure				
Cowlings	X	X		
Fairings	X	X		
Fiberglass Construction				
Monolithic				
Fairings	X	X	X	X
Covers	X	X	X	X
Doors		X		
Stiffened				
Fairings			X	
Covers			X	

TABLE 30. DEPOT SURVEY REPORTS OF SIGNIFICANT IN-SERVICE DAMAGE, UH-1 and AH-1 HELICOPTERS

Aircraft Component/Structure	Damage Modes							Description of Damage
	Disbond/ Delaminat'n	Dents/ Gouges	Punctures	Cracks	Fastener Damage	Buckling	Chafing	
ALUMINUM HONEYCOMB PANELS								
Main Beam - Base of Pylon (UH-1)	▨	▨						Delamination precipitated by internal corrosion,
Vertical Fin Panels - Tail	▨				▩		▩	Delamination precipitated by corrosion; fastener damage at blind holes; chafing by fiberglass fairings.
Fuel Cell Compartments Floors and Walls	▨						▧	Delamination primarily around inserts; buckling caused by hard landings.
Hydraulic Compartments Floors and Walls	▨							Delamination primarily around inserts.
Cargo Compartment Bulkheads (UH-1)	▧	▨						Heavy denting caused by cargo handling.
Engine Decks (Titanium Skins)	▧	▨	▨					Dents caused by foot traffic primarily; punctures by tool drops.
Service Decks	▧	▨	▨					Same as above.
Cabin Roof Structure	▧	▨	▨					Same as above

▨ Heavy Damage   ▩ Moderate Damage   ▧ Light Damage

TABLE 30 (Continued)

Aircraft Component/Structure	Damage Modes: Disbond/ Delaminat'n	Dents/ Gouges	Punctures	Cracks	Fastener Damage	Buckling	Chafing	Description of Damage
Fuel Filler Panels		■	▦					Dents and punctures caused by repeated impact by fuel nozzle.
Door Posts (UH-1)		■						Heavy denting caused by boot impacts in areas surrounding fuselage steps.
Floor Panels (UH-1)	▨	▨						Light denting and delamination caused by cargo impacts.
Main Beam (AH-1)	▦							Delamination precipitated by internal corrosion.
Ammunition Bay Floor (AH-1)	▦	■						Heavy denting caused by impacts by weapon stores during loading.
NOMEX HONEYCOMB PANELS								
Transmission Cowling (UH-1)	▨	▨	▦		▨			Handling damage. Nicks and gouges primarily at edges and corners.
Pylon Fairings, Fwd & Aft (AH-1)	▨	▨	▦		▨			Same as above.

■ Heavy Damage ▦ Moderate Damage ▨ Light Damage

TABLE 30 (Concluded)

Aircraft Component/Structure	Damage Modes Disbond/ Delaminat'n	Dents/ Gouges	Punctures	Cracks	Fastener Damage	Buckling	Chafing	Description of Damage
				FIBERGLASS COMPONENTS				
Vertical Fin Fairings			▨	▨	▨		■	Handling damage. Fiberglass chafes aluminum attaching members.
Tail Rotor Drive Shaft Covers			▨	▨	▩		■	Same as above.
Nose Cone (AH-1)			▩					Occasional damage caused by contact with work stands and equipment.
Nose Door (UH-1)			▨					Occasional damage caused by contact with work stands and equipment.
Fuselage Step Enclosures			▩		▨			Damaged by boot impact.

■ Heavy Damage ▩ Moderate Damage ▨ Light Damage

TABLE 31. DEPOT SURVEY REPORTS OF SIGNIFICANT IN-SERVICE DAMAGE, CH-47 HELICOPTER

Aircraft Component/Structure	Damage Modes							Description of Damage
	Disbond/ Delaminat'n	Dents/ Gouges	Punctures	Cracks	Fastener Damage	Buckling	Chafing	
ALUMINUM HONEYCOMB PANELS								
Fuel Pods	▨	▨	▨			░		Delamination precipitated by internal corrosion. Water enters through seams and punctures. Frequent ground handling damage.
Frames and Formers, Fuselage Tub Section	▨			░				Delamination precipitated by internal corrosion; heaviest in aft end of aircraft. Cracks caused by hard landings.
Floor Panels, Cockpit Passageway and Cargo Hook Wall	▨	▨	░					Panels heavily saturated with hydraulic oil. Dents and punctures caused by foot traffic and tool drops.
Cargo Ramp Hinge Cover		▨	░					Dents and punctures caused by movement of vehicles and heavy cargo over the ramp.
Roof Walkways	▨	▨	▩					Damage caused by foot traffic and tool drops.
Work Platforms (Fiberglass Skins)	▩	▨	▩	▩				Platforms exposed to oils and solvents. Dents and punctures caused by foot traffic and tool drops. Cracks at cable attachment.
Fuel Filler Panels	▨	▨	▩					Damage caused by repeated impacts with fuel nozzle.
Cargo Ramp Outer Shell	▨	▨	▩			▩		Much damage caused by lowering ramp onto ground objects. Buckling caused by operation with inner supports removed.

▨ Heavy Damage　　▩ Moderate Damage　　░ Light Damage

TABLE 31 (Concluded)

Aircraft Component/Structure	Damage Modes							Description of Damage
	Disbond/ Delaminat'n	Dents/ Gouges	Punctures	Cracks	Fastener Damage	Buckling	Chafing	
FIBERGLASS COMPONENTS								
Tunnel Covers			▦	▨	▦	▨	▦	Covers crushed and broken by foot traffic. Punctures caused by tool drops. Covers chafe aluminum attaching structure.
Fairings, Combining Gearbox and Short Shafts			▦	▨	▦	▨	▦	Same as above.
Crown Fairings, Forward and Aft Rotors					░		▨	Chafing caused by interference with rotating rain shields.
Rain Shields, Forward					░	▨		Rain shields warp and chafe against crown fairings.
Forward Transmission Aft Fairing		░	░		░			Handling damage. Dents and nicks primarily at edges and corners.
Fuselage Step Enclosures		░	░					Damage caused by boot impacts.

▨ Heavy Damage　▦ Moderate Damage　░ Light Damage

TABLE 32. DEPOT SURVEY REPORTS OF SIGNIFICANT IN-SERVICE DAMAGE, OH-58 HELICOPTER

Aircraft Component/Structure	Damage Modes							Description of Damage
	Disbond/ Delaminat'n	Dents/ Gouges	Punctures	Cracks	Fastener Damage	Buckling	Chafing	
ALUMINUM HONEYCOMB PANELS								
Floor Panels	■	▨	▨					Delamination precipitated by internal corrosion.
Radio Compartment Shelf	■	▨						Delamination heaviest in area of battery rack where battery fluid enters the panels around inserts.
Engine Decks	▨	■	■					Damage caused by foot traffic and tool drops.
Fuselage Panels, Aft of Passenger Doors	▨	■	■					Heavy damage caused by seat belt buckles when aircraft flown with belts hanging outside door.
Fuel Filler Panels	▨	■	■					Damage caused by repeated impacts with fuel nozzle.
FIBERGLASS COMPONENTS								
Engine Cowl Aft Fairing		▨	▨		▨		▨	Handling damage. Dents and nicks primarily at edges and corners. Minor chafing of metal attaching strips.
Transmission Cowl Forward Fairing		▨	▨		▨		▨	Same as above.

■ Heavy Damage ▩ Moderate Damage ▨ Light Damage

Of the several types of construction, bonded aluminum honeycomb panels suffer by far the highest rate of damage. Delamination caused by internal corrosion is the most prevalent type of failure with these components, and the problem is chronic in areas of some aircraft. Dents and punctures are other frequently occurring modes of damage with aluminum honeycomb construction. Areas subject to heavy foot traffic and tool drops are particularly vulnerable, as are areas subject to other types of impact such as cargo compartment bulkheads, protruding fuel pods and panels enclosing fueling ports.

Nomex/fiberglass construction is used in only a few applications and appears to hold up well in service. Only minor handling damage is reported. Fiberglass components also do well in service generally. The major problems occur when light structures are placed in areas where they can be stepped on and broken. Chafing of fiberglass against aluminum or other fiberglass is also a frequently reported problem. Minor handling damage and some fastener damage are the other types of reported problems with fiberglass components.

RELIABILITY FACTORS IN COMPOSITE STRUCTURES DESIGN

INHERENT RELIABILITY

The inherent modes of failure for aircraft structures are those arising from normal operations in the planned environment. For a military aircraft, which may be required to operate anywhere in the world, this encompasses a wide range of operating and environmental stresses. The aircraft structure will be designed to withstand the spectrum of flight and landing loads including high g-level maneuvers and hard landings. It will also be made survivable to combat damage and crash loads. Airframe fatigue lives are typically much in excess of the planned operating life of the aircraft, as witnessed by the many aircraft that are still operating well beyond their originally specified lives. With respect to environment, airframes are typically designed and qualified via structural and material testing to withstand extreme ranges of operation. This applies to both natural and induced environments, and includes factors such as temperature, moisture and salt atmosphere.

For composite structures, two modes of inherent failure might be anticipated: cracks and delamination, occurring either as the result of fatigue or from incipient flaws in materials or construction. Both of these modes should occur randomly and very infrequently. (Repetitive failures of this type in any one area of the fuselage would be indicative of a problem requiring design action.) Primarily, then, the reliability of composite structures will be a function of the rate of externally caused damage.

ENVIRONMENTAL HAZARDS

Figure 28 shows the significant environmental hazards to which composite structures may be exposed in service. Three types of environment are considered: (1) weather and climate, (2) operations and maintenance, and (3) combat. Each of the environmental hazards is related to the aircraft states and modes of operation in which it is most frequently encountered.

Hazards of the natural environment, those related to weather and climate, are relatively predictable and can be substantially neutralized through the selection of materials and the application of design allowables. Thus, if a composite material is known to be moisture-absorbing, and moisture content is known to have a degrading effect on strength or stiffness, the structure will be designed for the worst-case situation (maximum amount of absorbed moisture), particularly if it will be placed in a wet or humid environment. The same is true for the effects of solar radiation, extreme temperature, etc. It is of course impractical to design for every extreme of environment, and a structure exposed to baseball-sized hailstones or hurricane velocity winds could be expected to suffer damage. Conditions such as these are so rare that they can be dismissed in a general assessment of reliability, however.

This leaves as the only significant contributors to operational reliability hazards induced via operations, maintenance and combat. In this category

	Aircraft State/Flight Mode									
	Active							Inactive		
Environmental Hazard	Loading/ Unloading	Ground Running/ Taxi	Altitude Flying	Hover (IGE)/ NOE Flying	Landing	Ground Handling	Inspection and Maintenance	Parked/ Moored	Hangared/ Stored	
Weather/Climate										
Solar Radiation								X		
Extreme Temperature								X		
Humidity/Moisture								X	X	
Rain			X	X				X		
Snow								X		
Ice			X					X		
Hail			X					X		
Lightning			X					X		
Wind						X	X	X		Note 1
Operations/Maintenance										
Thermal Cycling/Shock		X	X	X						
Aircraft Fluids		X	X	X						
Vibration		X	X	X						
Airborne Particles/F.O.D.				X	X					
Foot Traffic	X						X			
Dropped Tool/Parts							X			
Dropped/Shifting Cargo	X		X		X					
Door Slamming	X						X			
Rough Handling						X	X			Note 2
Bird Strikes			X							
Impact with Terrain Objects				X	X					
Work Stands/Ground Vehicles						X	X	X	X	
Combat										
Ballistic Impacts			X	X						

Note 1: Adequately controlled via materials selection and design allowables

Note 2: Significant environmental hazards

Figure 28. Environmental Hazards Related to the Aircraft States and Flight Modes in Which They are Most Frequently Encountered

also some hazards can be controlled effectively by design; thermal cycling and exposure to aircraft fluids are two of these. Knowing beforehand that materials will be placed in an engine compartment or hydraulics bay allows the designer to compensate for the degradation in properties that these environments may produce.

IMPACT DAMAGE

It is concluded from the foregoing and from the surveys of service experience reported on earlier, that from the standpoint of reliability in service, the significant concern in the design of composite structures for helicopters will be damage caused by impact. The assumption applies of course to composite structures at a mature stage of development. The first of the structures to be introduced to service may have some inherent deficiencies that surface in the form of early reliability problems.

The view that the reliability of composite structures will be predominantly a function of exposure to impact is consistent with the findings of the service experience study. The surveys of Army depots disclosed that with the exception of corrosion of aluminum honeycomb, almost all of the damage to these kinds of structures occurs as a result of some type of impact. And the Air Force study of advanced composite structures is also concentrating entirely on impact damage (Reference 5).

Figures 29 through 32 illustrate areas of the helicopter airframe that are particularly vulnerable to various types of impact damage as determined by the service experience surveys. Later in this report it will be shown how this information is used to assess the potential reliability of advanced composite structures concepts.

Types and Degrees of Impact Damage

The damage sustained by a structure subjected to impact involves a large number of variables, including:

Impacting Object

Shape (blunt or sharp)

Incidence of impact (direct, glancing, etc.)

Location of impact* (center/edge)

Impact energy

*Deflection at center of panel produces less damage at a given energy level.

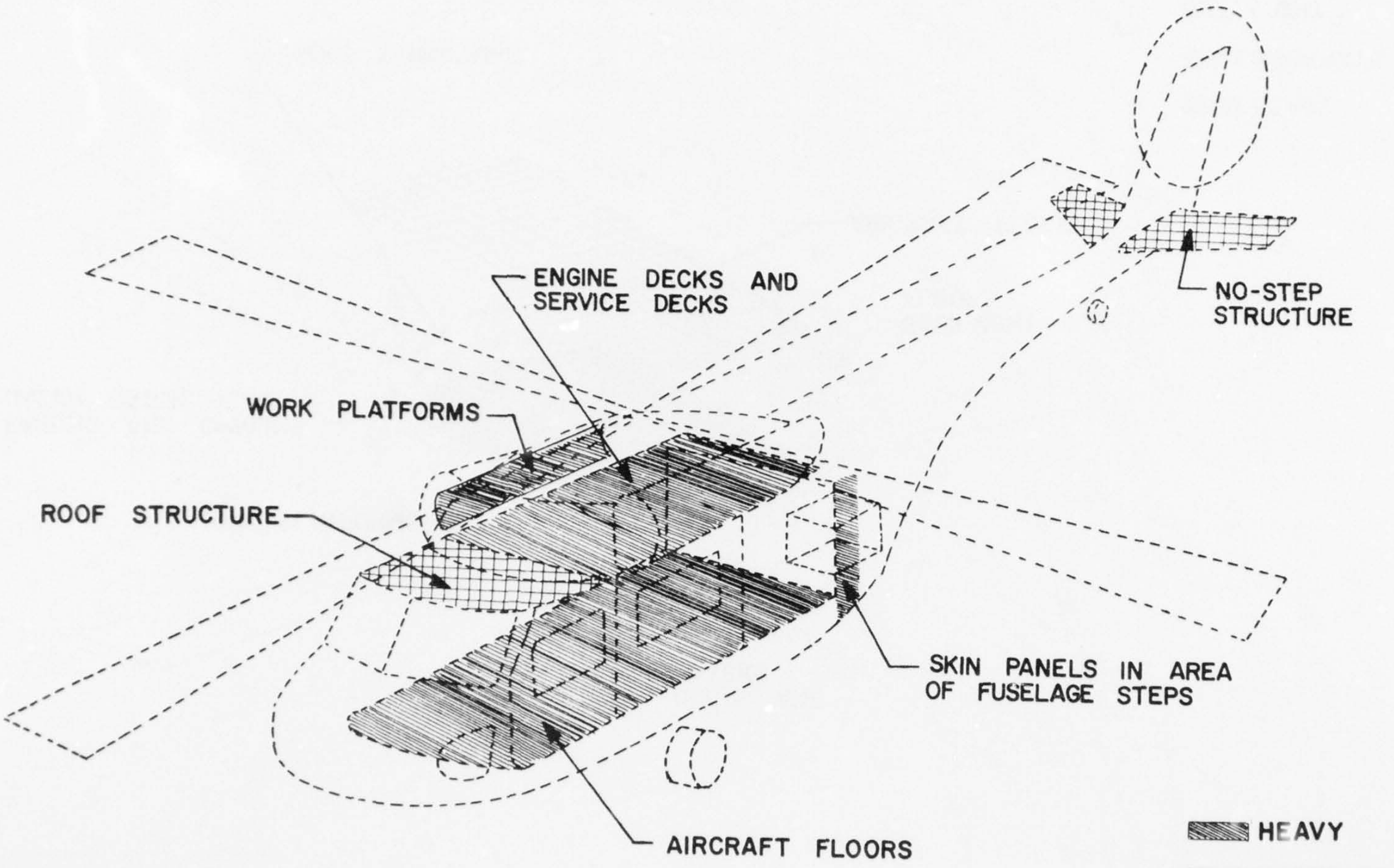


Figure 29. Typical Areas of Structure Vulnerable to Damage by Foot Traffic

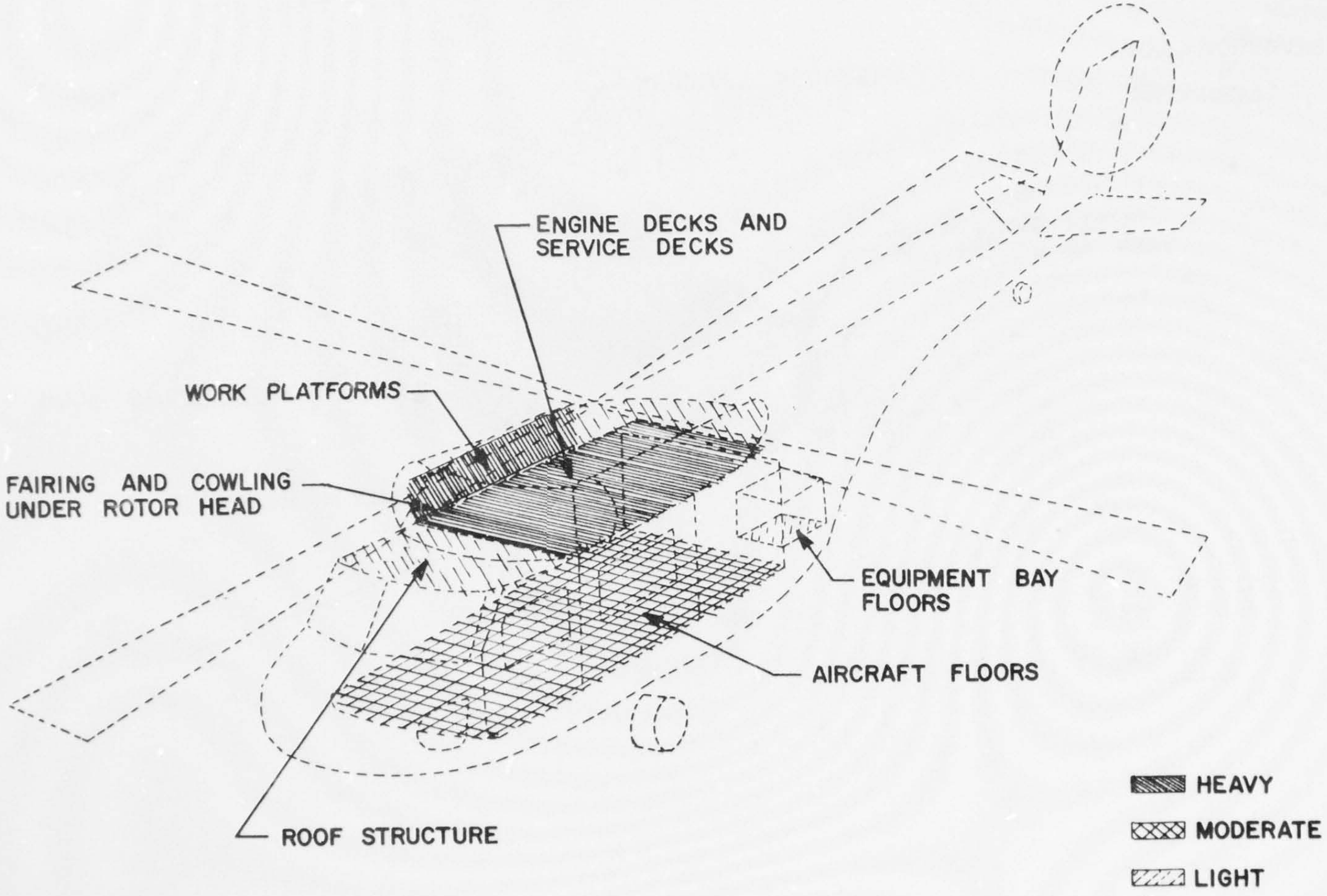


Figure 30. Typical Areas of Structure Vulnerable to Dropped Tools and Parts

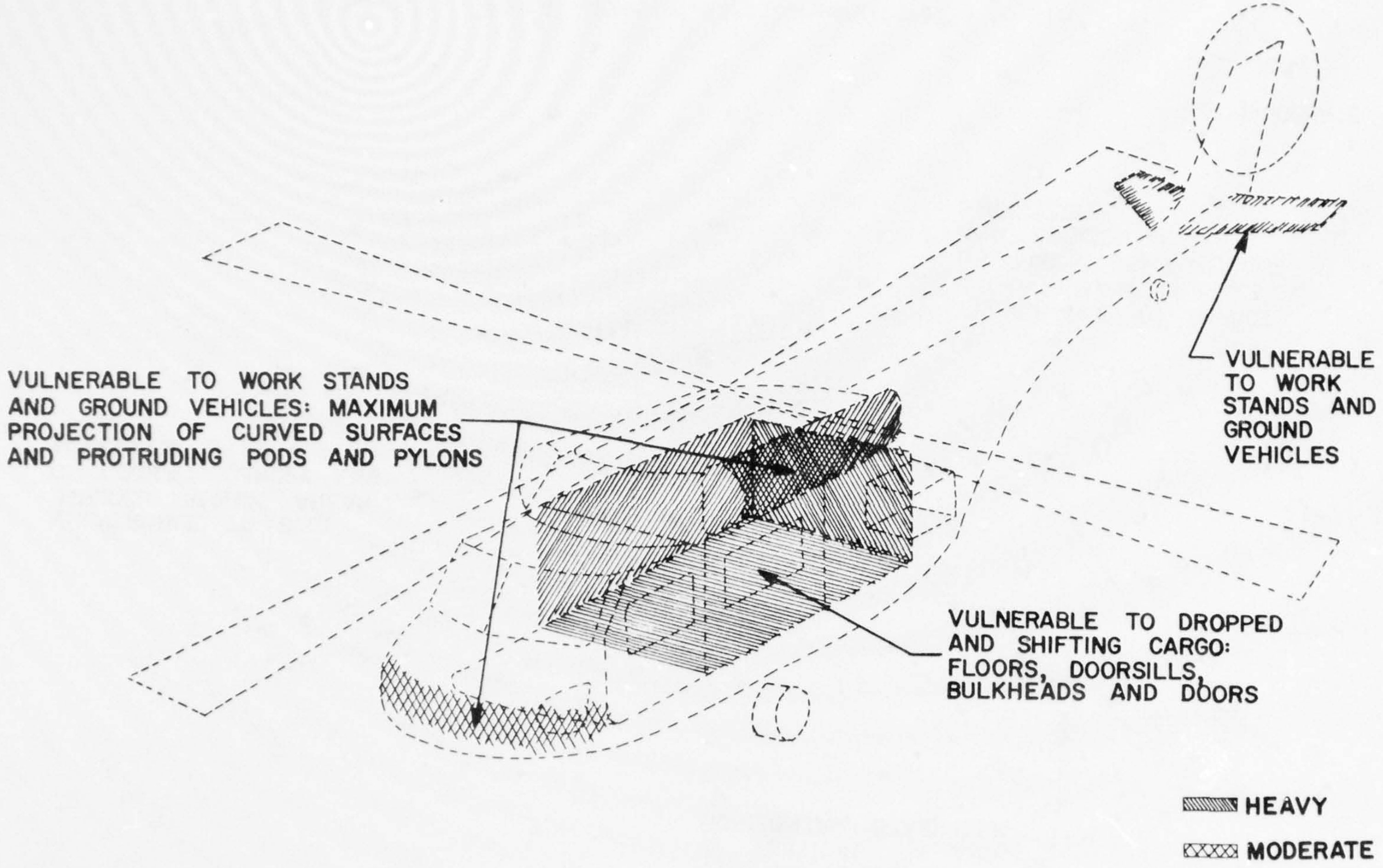


Figure 31. Typical Areas of Structure Vulnerable to Dropped and Shifting Cargo and Impact by Ground Vehicles

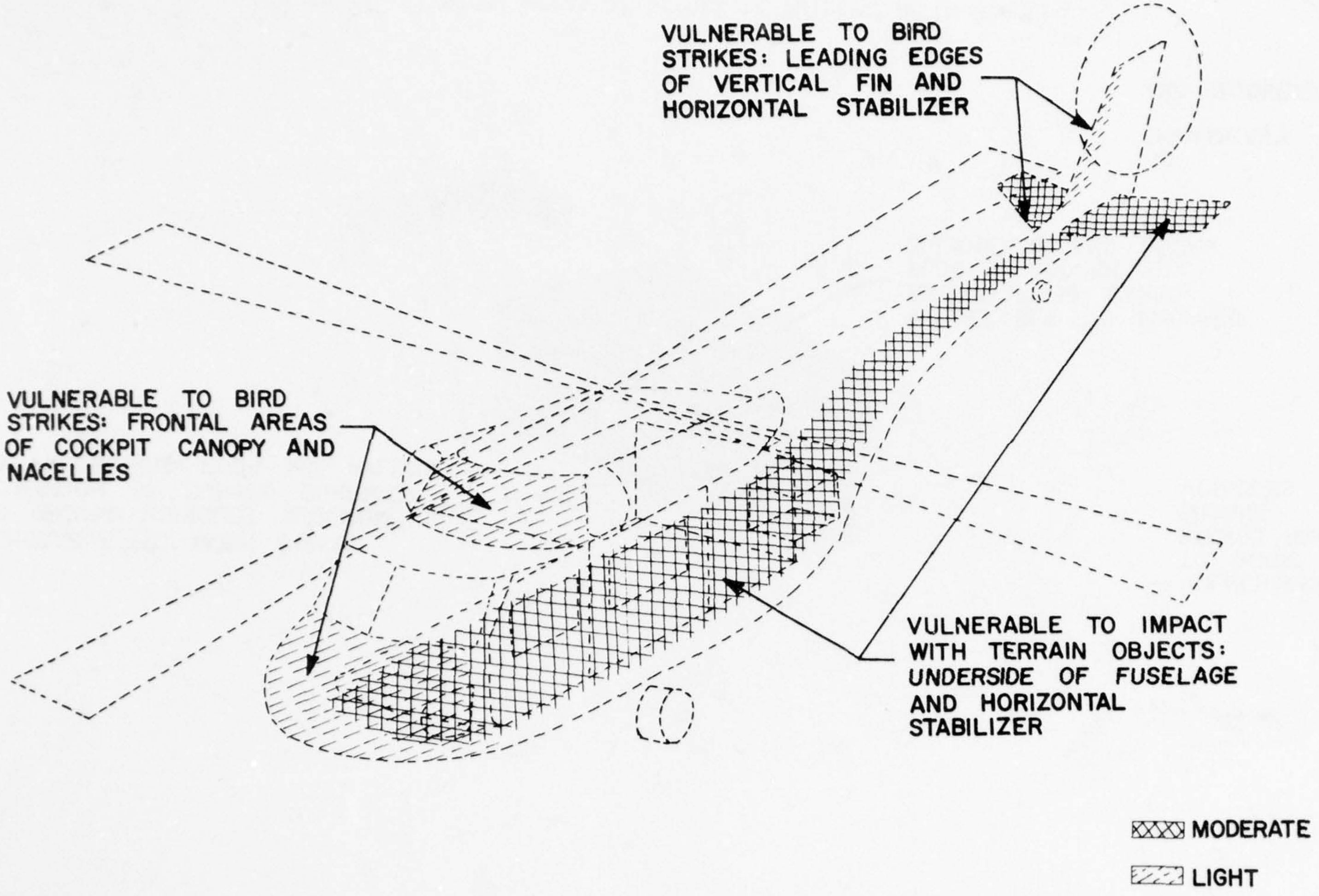


Figure 32. Typical Areas of Structure Vulnerable to Bird Strikes and Impact with Terrain Objects

Design of the Structure

Type of material (properties of fibers/matrix)

Material form (unidirectional/woven)

Type of construction (monolithic/sandwich)

Material thickness

Ply orientation

Edge restraint

Presence of doublers, stiffeners, etc.

All of these variables will affect the type of damage sustained by a composite structure subjected to a single impact. The damage itself is a variable possessing certain characteristics, namely:

Type (dent, crack, puncture, etc.)

Size (area, depth)

Criticality (negligible, repairable, etc.)

Location (surface/subsurface)

When all of these variables are considered together, it is clear that a given composite structure has the potential of being damaged in a great many different ways. The reliability of the structure will depend not only on the types of damage it receives but also on the frequency of damage. This introduces another set of variables involving the mission of the aircraft, the environment in which it operates and the quality of maintenance it receives.

MATERIAL AND DESIGN FACTORS

Material Factors

Each material possesses mechanical properties which make it more or less vulnerable to various types of damage. High interlaminar shear strength reduces a material's susceptibility to delamination. High compression strength provides protection against crushing. Other properties affect the resistance of the material to other types of damage.

Table 33 lists some of the principal mechanical properties of composites and aluminum. The table was assembled by Sikorsky's Structures and Materials Branch from published sources (References 6 through 14) and from data developed through in-house test programs.

With two exceptions the composite properties are based on a particular laminate configuration and thickness, one that might be used for an aircraft skin. It is important to note that other configurations and thicknesses would substantially alter many of these properties.

---

[6] ADVANCED COMPOSITES DESIGN GUIDE, VOLUME IV, MATERIALS, Third Edition, Advanced Development Division, Air Force Materials Laboratory, Wright-Patterson Air Force Base, Ohio, January 1973.

[7] KEVLAR 49 DATA MANUAL, E. I. DuPont DeNemours and Company, Wilmington, Delaware.

[8] MIL-HDBK-5B, METALLIC MATERIALS AND ELEMENTS FOR AEROSPACE VEHICLE STRUCTURES, Department of Defense, September 1971.

[9] SCOTCHPLY PRODUCT INFORMATION, SP-114, Industrial Specialities Division, 3M Company, St. Paul, Minnesota.

[10] FLIGHTWORTHY GRAPHITE FIBER REINFORCED COMPOSITES, VOLUME 3, Northrop Corporation, Report Number AFML-TR-70-207, U.S. Air Force Materials Laboratory, Wright-Patterson Air Force Base, Ohio, October 1970.

[11] Flonc, N., CHARACTERIZATION OF BORON, GRAPHITE AND GLASS FILAMENT/ ORGANIC MATRIX COMPOSITE MATERIALS, Sikorsky Report Number SER-50644, Sikorsky Aircraft Division, Stratford, Connecticut, January 1970.

[12] SIKORSKY STRUCTURES MANUAL, Sikorsky Aircraft Division, Stratford, Connecticut.

[13] MATERIAL PROPERTIES OF HEXCEL HONEYCOMB MATERIALS, TSB 120, Hexcel Corporation, Dublin, California, 1975.

[14] MIL-HDBK-17A, PLASTICS FOR AEROSPACE VEHICLES, PART I, REINFORCED PLASTICS, January 1971.

TABLE 33. PROPERTIES OF COMPOSITES AND ALUMINUM

Property/Characteristic	Material					Composite Laminate Configuration
	Boron/ Epoxy	Kevlar/ Epoxy	Graphite/ Epoxy	Fiberglass/ Epoxy	Aluminum 2024-T3	
Tensile Strength (ksi)	95 $\boxed{6}$	92 $\boxed{7}$	90 $\boxed{6}$	75 $\boxed{9}$	65 $\boxed{8}$	A
Tensile Elongation to Failure (%)	5 6	2 $\boxed{T}$	1 $\boxed{T}$	2 $\boxed{T}$	15 $\boxed{7}$	A
Tensile Modulus (psi x 10^6)	17.3 $\boxed{6}$	5.6 $\boxed{7}$	9.2 $\boxed{6}$	3.7 $\boxed{9}$	10.5 $\boxed{8}$	A
Compression Strength (ksi)	166 $\boxed{6}$	29.5 $\boxed{7}$	90 $\boxed{6}$	75 $\boxed{9}$	40 $\boxed{8}$	A
Strain Energy (in-lb/in^3)	261 $\boxed{C}$	756 $\boxed{C}$	440 $\boxed{C}$	760 $\boxed{C}$	8,237 $\boxed{C}$	A
Interlaminar Shear Strength; Individual Minimum (psi)	13,000 $\boxed{T}$	4,500 $\boxed{T}$	13,000 $\boxed{T}$	7,500 (typical) $\boxed{T}$	40,000 (typical) $\boxed{7}$	A
Shear Strength Perpendicular to Laminate Plane (ksi)	96 $\boxed{6}$	28 $\boxed{7}$	38 $\boxed{11}$	30 $\boxed{T}$	40 $\boxed{8}$	C
Impact Strength (ft-lb/in^2)	40 $\boxed{2}$	150 $\boxed{7}$	20 $\boxed{7}$	275 $\boxed{7}$	220 $\boxed{7}$	A
Fracture Toughness (ksi - in $^{1/2}$)	N/A $\boxed{2}$	23 $\boxed{7}$	22 $\boxed{7}$	14 $\boxed{7}$	73 (2024-T4) $\boxed{7}$	B
Transverse Compression Strength (ksi)	45 $\boxed{6}$	20 $\boxed{7}$	31 $\boxed{6}$	20 $\boxed{9}$	40 $\boxed{8}$	A
Barcol Hardness	40-100 $\boxed{7}$	40-45 $\boxed{7}$	50-55 $\boxed{7}$	70 $\boxed{9}$	120 Brinell $\boxed{7}$	A
Crack Propagation $F = (1/2\sigma\ 2/E) / K_c$	0 $\boxed{C}$	.033 $\boxed{C}$	.020 $\boxed{C}$	.054 $\boxed{C}$	.113 $\boxed{C}$	A
Buckling Tolerance ($E\ x\sigma_c$)	2,872 $\boxed{C}$	165 $\boxed{C}$	828 $\boxed{C}$	278 $\boxed{C}$	420 $\boxed{C}$	A
Bearing Strength (ksi)	135 $\boxed{7}$	40 $\boxed{7}$	130 $\boxed{6}$	47 $\boxed{14}$	114 $\boxed{8}$	A

Laminate Configuration

A. 0°/90° Crossply; .040 thick;
$V_f = 60\%$

B. 0°/90°/±45°

C. ± 45°

$\boxed{\quad \boxed{\ }}$ Reference Source
T = Sikorsky Aircraft Test Data
C = Calculated Value

Table 34 lists the mechanical properties for core materials. Here a typical density has been selected, and just as the properties of composites vary with laminate configuration and thickness, some properties of the core materials would change substantially if other densities were used.

TABLE 34. PROPERTIES OF CORE MATERIALS

Property/Characteristic	Material		
	Aluminum Honeycomb	Nomex Honeycomb	Structural Foam
Density (lb/ft^3)	6 $\boxed{13}$	6 $\boxed{13}$	35 $\boxed{12}$
Compression Strength (psi)	680 $\boxed{13}$	825 $\boxed{13}$	6,000 $\boxed{12}$
Specific Compressive Strength (inches)	.066 $\boxed{C}$	.079 $\boxed{C}$	.099 $\boxed{C}$
Shear Strength (psi)	455 $\boxed{13}$	260 $\boxed{13}$	1,800 $\boxed{12}$
Elastic Limit (%)	0.3 $\boxed{13}$	1.4 $\boxed{13}$	2.3 $\boxed{12}$
Yield Point (Yes/No)	Yes ☐	No ☐	No ☐

☐ Source Reference
C = Calculated Value

Basic mechanical properties were used in part to establish damage tolerance ratings for aluminum, the three commonly used composite materials (fiberglass, Kevlar and graphite) and the three commonly used core materials (aluminum honeycomb, Nomex honeycomb and structural foam). The properties used as aids to developing damage tolerance ratings are given in Table 35. The ratings are summarized as an element of the R&M assessment technique described later in this report.

TABLE 35. DAMAGE TOLERANCE RATING FACTORS

	Damage Tolerance Rating Factor	
Damage Mode	Composites/Aluminum	Core Materials
Abrasion	Barcol Hardness	
Denting	Yield Point	Elastic Limit
Puncture	Shear Strength Perpendicular to Laminate	
Delamination	Interlaminar Shear Strength	
Cracking	Strain Energy Impact Strength	Yield Point (Yes/No)
Fastener Damage	Bearing Strength	
Crushing	Compressive Strength	Specific Compressive Strength
Buckling	Buckling Tolerance	

Design Factors

In addition to the mechanical properties of the materials, characteristics of the design may affect damage susceptibility and damage tolerance, and hence the reliability of the structure in service. Table 36 describes the key design factors having a potential effect on structural reliability, either positive or negative. Later in this report, these factors and others are used to develop an R&M assessment technique for advanced structures design concepts.

TABLE 36. RELIABILITY DESIGN FACTORS

Design Factor		Effect on Reliability
Construction Form	Monolithic Sheet	More flexible than sandwich; greater impact strength.
	Stiffened Sheet	Similar to monolithic sheet.
	Sandwich	Facings thinner than equivalently loaded monolithic panels; more easily punctured. Less impact resistant than monolithic sheet. Bond failures may occur between core and facings due to overstress or impact.
Stiffener Form	Open Section	Less stable than closed section forms; more vulnerable to twisting or buckling type failures.
	Hollow Core	Closed section more stable than open section; less vulnerable to buckling or twisting type failures.
	Foam Core	Similar to hollow core.
Method of Assembly	Co-cured	Excellent bond strength due to resin intermixing.
	Adhesive Bond	Simple structural joint. Cleanliness and quality control critical to achieving structural integrity.
	Mechanical Fasteners	May loosen and cause fretting or separation of joint.
Contour	Double Curvature/ Wrapped Surface	Sharp exposed radii may be vulnerable to impact.
	Flat Surface	Least vulnerable to impact.
Accessibility	Restricted	Inability to inspect properly may allow flaws or damage to progress to advanced stages.
Load Intensity	Lightly Loaded	Damage has minimal effect on structural integrity; adjacent structure supports load in event of localized damage. Most easily damaged due to lightweight construction.
	Moderately Loaded	Structural integrity more seriously affected by damage.
	Heavily Loaded	Any damage is critical.
Interface Constraints	Equipment mounting provisions and cutouts.	Local structure reinforcement for equipment adds to complexity; introduces potential failure modes. Affected by loads existing in structure and introduced at interface.

MAINTAINABILITY FACTORS IN COMPOSITE STRUCTURES DESIGN

DESIGN FACTORS

The maintainability of an airframe structure is a measure of the ease with which it can be inspected, repaired, and if a separable part of the airframe, replaced. Although static in nature, airframe structures may possess characteristics that tend either to enhance or degrade maintainability. Some of these characteristics are generic to the type of structure while others vary with the particular design. Table 37 enumerates the significant design factors affecting maintainability and describes the nature of these effects. Figures 33 through 36 illustrate key factors. The R&M assessment technique presented later in this report incorporates an evaluation of these design factors.

INSPECTION

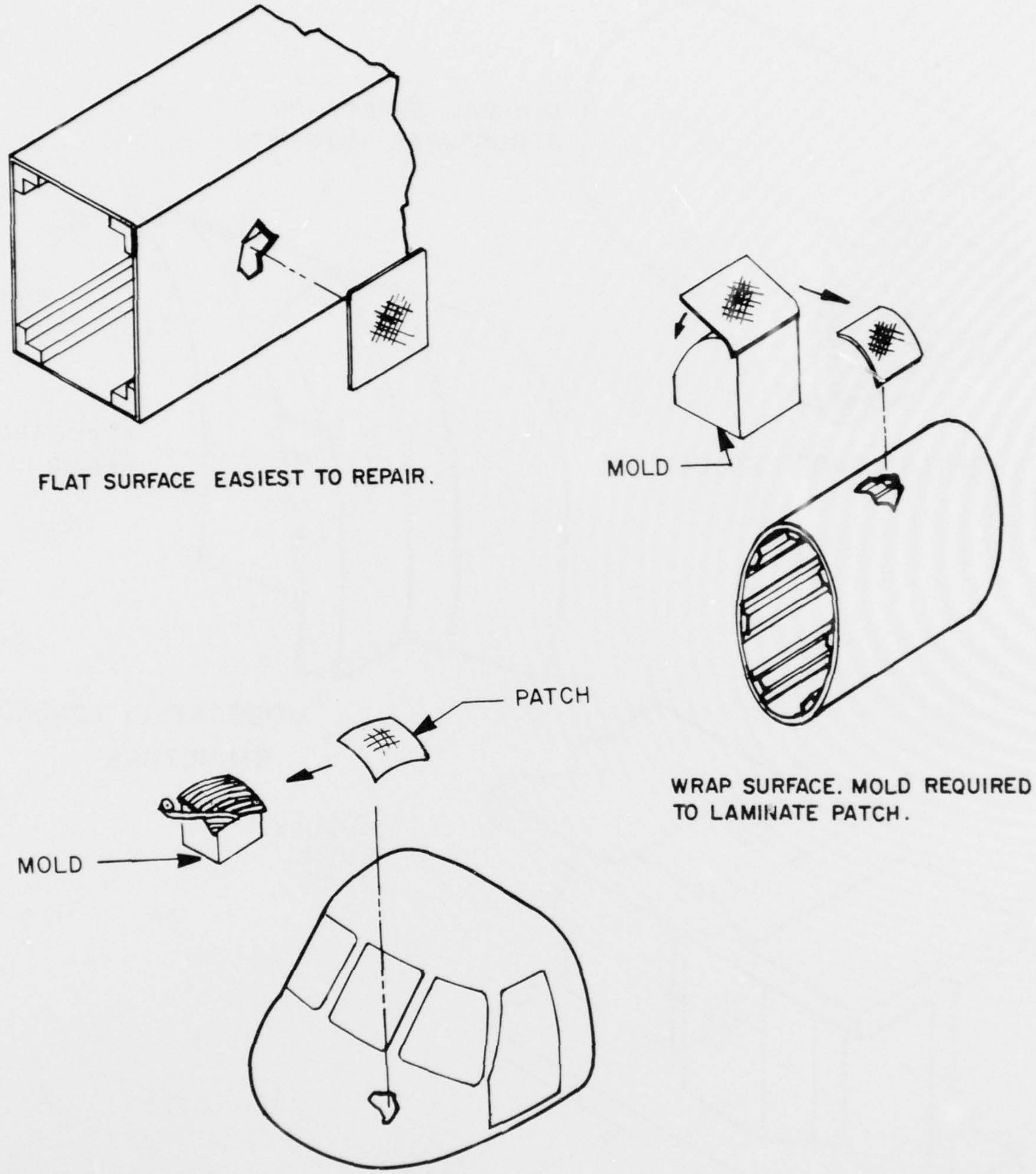
Composite materials, unlike metals, do not yield under stress. Although superior in strength to metals in many applications, the stress-strain curve for composites is essentially a straight line to fracture. This property, coupled with the laminated construction of composites, presents problems for inspection. A metal structure subjected to overstress or severe impact will normally exhibit visible damage at the surface in the form of cracks, dents or structural deformation of some type. This may not be true for a composite structure. Because of its elasticity, a composite subjected to impact will tend to resume its natural shape (unless the impact is severe enough to cause fracture). The impact, while producing no surface damage, may create shear stresses large enough to cause internal delamination. Although exhibiting no physical evidence of damage, the structure may have in fact begun to fail.

Presently, for the few composites now in service the primary method of inspection is audio sonic (coin tapping). Even at the depots, where more advanced techniques such as ultrasonics are available, coin tapping is the method most preferred. In the course of the surveys conducted under this program, depot personnel reported that ultrasonics is a more complicated and time-consuming method of inspection and that it generally produces no better results. Because of their large cross sections, rotor blades are the one component for which ultrasonic techniques have been found to be more effective than coin tapping.

Although coin tapping is considered a reliable method today, its use has been confined to the inspection of relatively simple, noncritical structures, primarily aluminum honeycomb panels which produce distinctive differences in sound in areas where voids or delaminations are present. Future aircraft will contain highly loaded primary structures comprised of thick laminate buildups, and areas of these structures may be relatively inaccessible to inspection. Coin tapping will probably not be an effective method of inspection for such structures.

TABLE 37. MAINTAINABILITY DESIGN FACTORS

Design Factor		Effect on Maintainability
Construction Form	Monolithic Sheet	Repairability good when both sides of panel exposed. Simple, well-established repair procedures.
	Stiffened Sheet	Presence of stiffeners makes repair more complex.
	Sandwich	Bond failures between core and facings may be difficult to detect. Repairability generally good; damaged core can be filled in and patched over. Absence of complex shapes and curvatures simplifies repair.
Stiffener Form	Open Section	Easiest to repair because all surfaces are exposed.
	Hollow Core	Repair limited to external surfaces because of inaccessibility to interior.
	Foam Core	May offer slight advantage over hollow core since core material can be filled-in to provide a mold for cure-in-place repair.
Method of Assembly	Co-cured	Joint is permanent; must be cut apart for repair.
	Adhesive Bond	Absolute cleanliness required to achieve good bond; difficult to implement in field environment. Verification of integrity of repair difficult under field conditions. Some adhesives require refrigeration and have limited shelf life. High skill required.
	Mechanical Fasteners	Easiest type of joint to disassemble. Caution needed in use of mechanical fasteners for repair to avoid introducing stress concentrations and to avoid incompatibility of materials (aluminum and graphite for example).
Contour	Double Curvature	Material must be stretched or shrunk to conform to 3-dimensional surfaces; special molds required. Labor to laminate contoured parts related to amount of curvature.
	Wrapped Surface	Less difficult to laminate than double curvature: mold required.
	Flat Surface	Easiest to repair; no molds required.
Accessibility	Restricted	Poor accessibility impedes inspection. Restricted access impedes on-aircraft repairs; limits the use of equipment; increases the probability of faulty repair; adds to repair time.
Load Intensity	Lightly Loaded	Quality of repair less critical than more heavily loaded structures; visual inspection of repair adequate.
	Moderately Loaded	Quality of repair is important; verification of integrity via non-destructive inspection techniques may be necessary.
	Heavily Loaded	Quality of repair is critical; usually requires replacement or custom-engineered repair. Verification of integrity via non-destructive inspection techniques will be necessary.
Interface Constraints	Equipment mounting provisions and cutouts.	Requirements for equipment interchangeability impose dimensional constraints on repair (flush surfaces for example).



FLAT SURFACE EASIEST TO REPAIR.

WRAP SURFACE. MOLD REQUIRED TO LAMINATE PATCH.

COMPOUND CURVATURE. MOST DIFFICULT REPAIR. MATERIALS MUST BE LAID UP IN STRIPS.

Figure 33. Effect of Contour on Repair

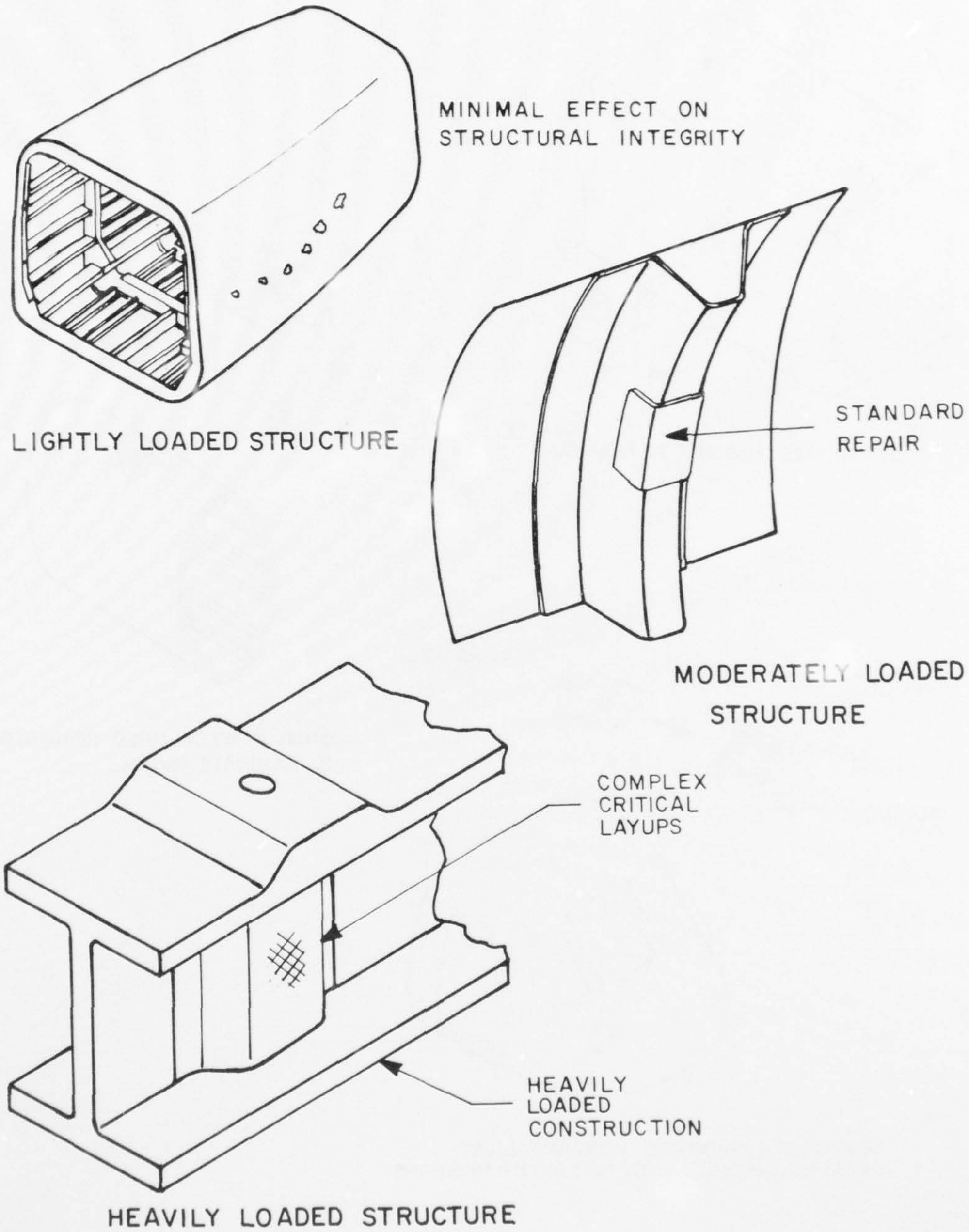
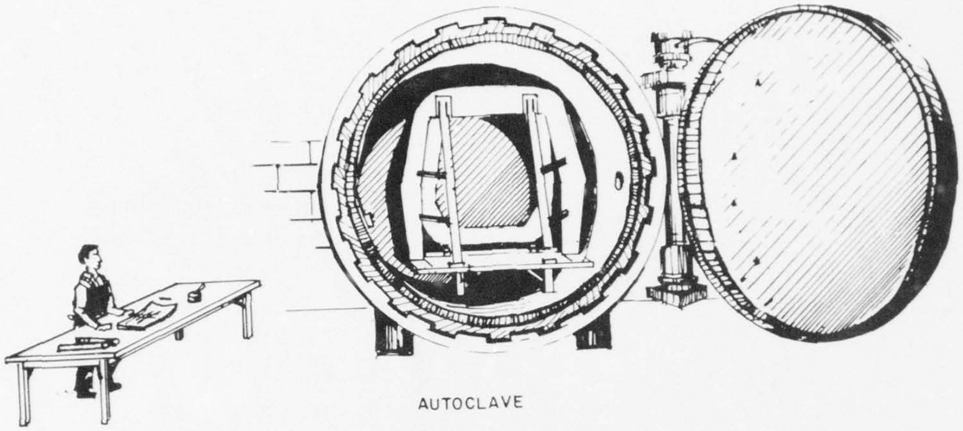
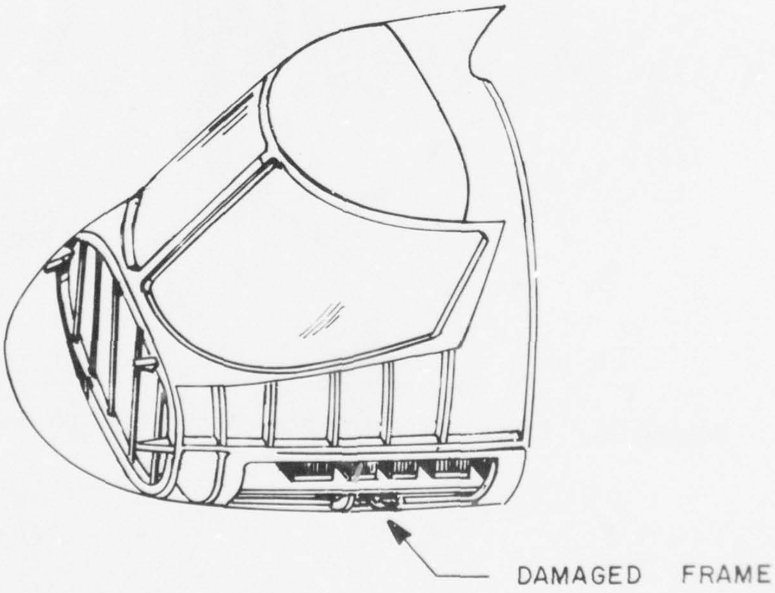


Figure 34. Effect of Load Intensity on Repair



IDEAL REPAIR ENVIRONMENT

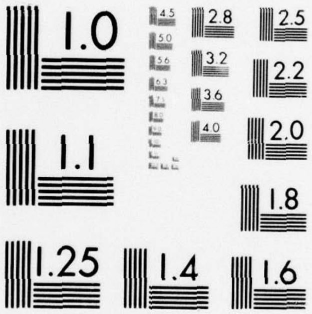


CONSTRAINED AREA REPAIR

Figure 35. Effect of Accessibility on Repair

AD-A077 373 UNITED TECHNOLOGIES CORP STRATFORD CT SIKORSKY AIRCR--ETC F/G 15/5
ADVANCED STRUCTURES CONCEPTS R AND M/COST ASSESSMENTS.(U)
SEP 79 T N COOK, B F KAY DAAJ02-77-C-0061
UNCLASSIFIED SER-510016 USARTL-TR-79-18 NL

2 OF 3
AD
A077373



MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS-1963-A

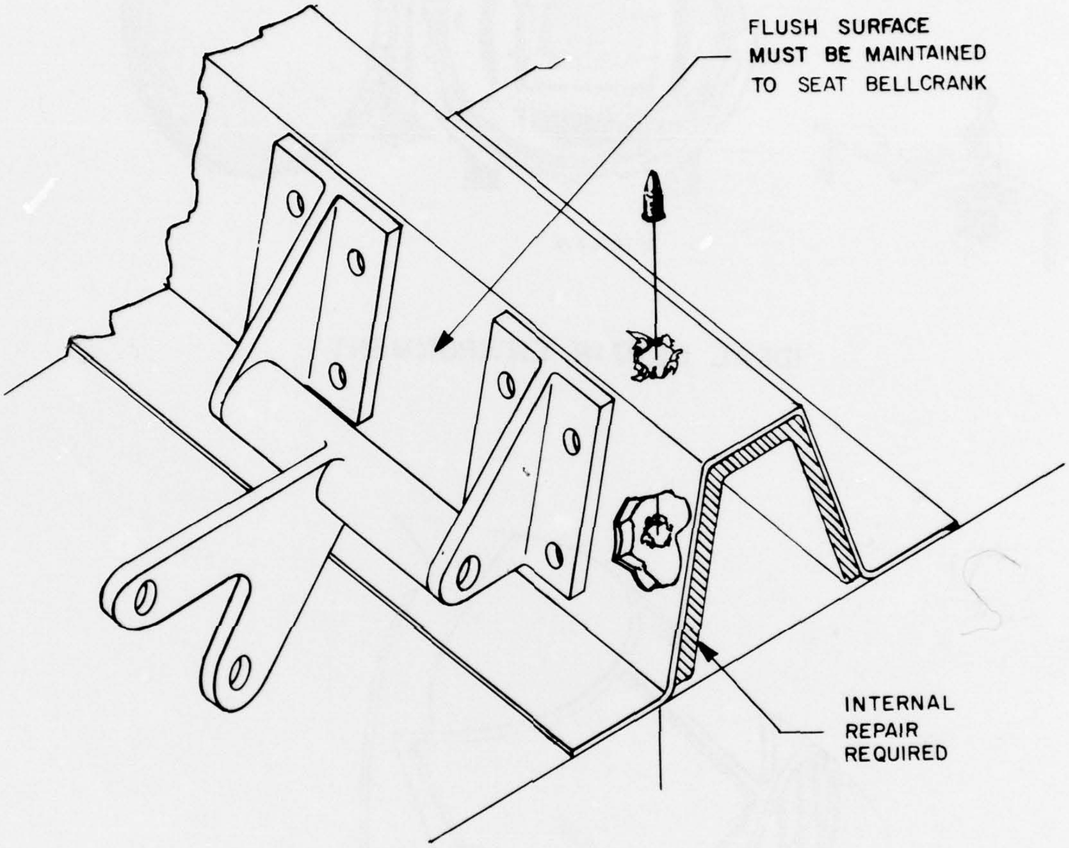


Figure 36. Effect of Interface Constraints on Repair

Besides ultrasonics, the other nondestructive method of inspection that might be considered for composites is radiography. Like ultrasonics, radiography involves the use of complex equipment and very specialized skills and appears highly unsuited to the Army field environment. Further development work will be needed to provide an effective inspection capability for advanced composite structures, either through design or through improved maintenance skills and equipment.

REPAIR

Techniques for repair of advanced composite structures are being investigated under a number of R&D programs with the DoD. Most of this work to date has been done within the fixed-wing community, where development of advanced composite structures has been most active. Within the helicopter industry, repair of advanced composites has been confined largely to rotor blades, the first components to use these materials for primary structure on a large scale.

Concepts employing advanced composites are now being proposed for many areas of primary structure in the helicopter airframe. Little of this work has progressed beyond the conceptual stage, and the specific form and details of these designs are not yet established. Nothing substantial has been done with regard to repair of these structures.

Repair assumes increasing importance for structures that are integral parts of the airframe. Since integral structures tend to be complex and large, they are also the most difficult to repair. This presents something of a paradox for R&M. The structures that are easiest to repair (small fiberglass fairings for example) are the ones for which repairability is least crucial, since they are relatively inexpensive and easy to replace. The structures that will be most difficult to repair (transmission support beams for example) are the ones for which repairability is most crucial, since they are expensive and very difficult or impossible to replace. The importance of repairability is also related to the expected frequency of damage, of course.

Techniques for repair of light to moderately loaded monolithic panels and sandwich panels, either flat or single curvature, are already well developed. The procedures are relatively simple, require only average skills and are suited to the field environment. Largely conceptual at this point are techniques for repair of heavily loaded structures such as frames and beams and panels with double curvature surfaces. It is expected that these techniques, when developed, could be relatively complex, require substantial skill, and may not be suited to the field environment. Considerable work in this area remains to be done.

The design of repairs for composite structures will have to satisfy certain criteria related to strength and durability, functional performance and technical feasibility. Some of the more significant of these are:

1. Restoration of structural strength and stiffness.

2. Restoration of finish and special surface treatments (wire mesh or conductive paint for lightning protection, for example).

3. Restoration of (minimal change in) aerodynamic contour where applicable.

4. Minimal weight increase.

5. Use of repair materials that are mechanically and chemically compatible with the parent structure (avoidance of aluminum rivets in graphite, for example).

6. Use of repair materials that are compatible with the temperature environment of the parent structure.

7. Use of mechanical fasteners only when the laminate characteristics of the repair material and parent structure permit.

8. Preservation of the functional characteristics of the parent structure (avoidance of interferences, etc.).

9. Avoidance of thickness changes that reduce or prevent fastener engagement.

10. Avoidance of erosion, edge peeling and other forms of repair deterioration.

11. Ability to verify the structural integrity of repair via test or inspection.

12. Repair techniques, materials and equipment that are compatible with the Army field environment.

13. For the combat environment, rapid restoration to flight status via quickly performed (permanent or interim) repairs.

Table 38 and Figures 37 through 41 describe general types of repairs for composite structures. The figures were taken from References 15 and 16. The conditions under which each type of repair might be used are stated and comments are made relative to known limitations and constraints on their use. Table 39 relates types of repair to generic types of damage.

---

[15] Foreman, C., McGovern, S. A., and Knight, R., S-34 GRAPHITE/EPOXY SPOILER FABRICATION OF TEN SHIPSETS AND DAMAGE REPAIR STUDY, Vought Corporation Systems Division, Report No. NADC-76234-30, Naval Air Development Center, Warminster, PA, May 1976.

[16] LaSelle, R. M., REPAIR PROCEDURES FOR ADVANCED COMPOSITE STRUCTURES, VOL. II, REPAIR GUIDE, General Dynamics Corporation, Report No. AFFDL-TR-76-57, Volume 2, U. S. Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio, December 1976.

Table 38. REPAIR METHODS

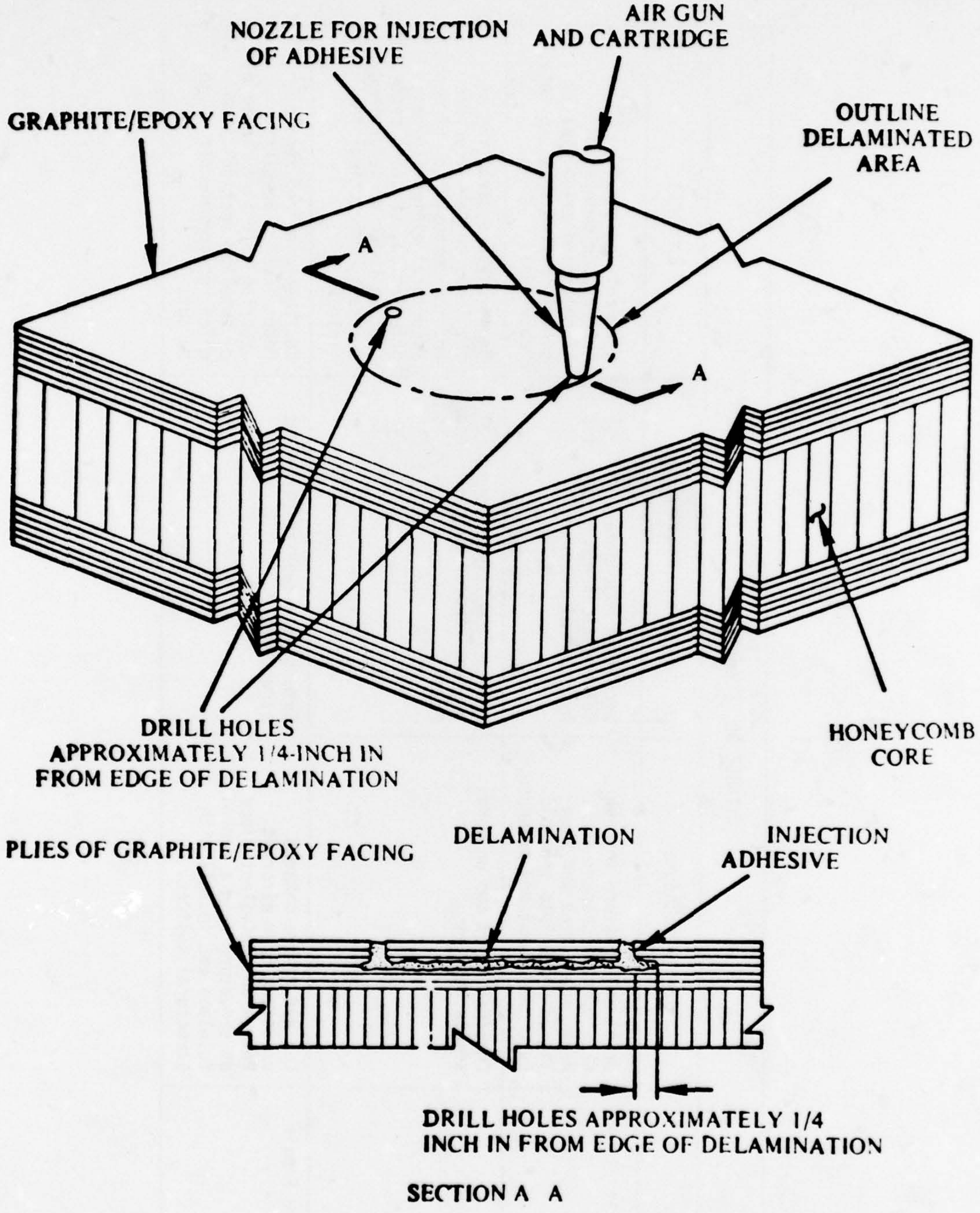
Type Repair	Basic Procedure	Damage Description	Comments
Surface Touch-up	Fill and blend surface defects using resin filler material and epoxy paint.	Minor localized surface damage such as nicks, chips, scratches and abrasions.	Not a structural repair.
Surface Refinishing	Remove loose or flaking paint. Clean and lightly abrade the surface using lacquer thinner and abrasive paper. Spray paint the surface using epoxy primer and acrylic lacquer.	Flaking, peeling, pitting and minor abrasion of protective finish.	Not a structural repair.
Cure-in-Place Patch Repair (Surface or Flush)	Remove damaged skin and clean surface. Cut to size and lay up pre-preg laminates on parent structure. Cure in place using heat and pressure.	Cracks, cuts, tears and punctures of skin panel or structural member where the shape and/or contour of the structure prohibits use of a pre-cured patch.	Layup comprised of pre-preg composite or titanium foil laminates and epoxy film adhesive. Ply stacking and orientation must match parent structures. Special storage of materials and high skill required.
Prefabricated Patch or Doubler	Remove damaged skin and clean surface. Install prefabricated patch using a film or paste epoxy adhesive and heat and pressure.	Cracks, cuts, tears and punctures of a panel or structural component where a standard patch size can be used regularly or an irregular shape must be accommodated and a cure-in-place repair is not feasible.	Factory fabricated to match part contour, ply count and laminate orientation. Layup consists only of patch and adhesive. Less skill required. Practical only for relatively flat panels and/or frequently damaged parts.
Potting Compound Repair (Sandwich Panels)	Remove damaged skin and carve out a cavity in the core, creating an undercut around the skin cutout. Pack potting compound into the cavity and cure under heat and pressure. Sand flush with skin and install skin patch over repair.	Small diameter tear or puncture in honeycomb or foam core sandwich panel causing core damage in excess of that allowed for simple skin patch repair.	Creating skin undercut may be troublesome. Potting compound may tend to pack into honeycomb cells. Heavier than honeycomb plug. Requires two operations: plug and skin patch. More difficult than prefabricated plug/patch.

TABLE 38 (Continued)

Type Repair	Basic Procedure	Damage Description	Comments
Core Plug Repair	Remove damaged skin and carve out a cylindrical cavity in the core. Cut a core plug from honeycomb material, fit to cavity, and bond in place with a wafer in the floor of the cavity using paste epoxy adhesive. Cure under heat and pressure. Install skin patch over core plug.	Large diameter tear or puncture in honeycomb sandwich panel in excess of that permitted for potting compound repair. Used where prefabricated plug/patch unavailable.	Forming plug from honeycomb stock may be difficult with field tools. Requires two operations: plug repair and skin patch. More difficult than prefabricated plug/patch.
Prefabricated Plug/ Patch Repair	Remove damaged skin and core by carving out a cylindrical cavity *in the sandwich structure,* sized to accommodate a standard plug/patch. Prepare skin surface surrounding cavity and bond plug/patch in place with a wafer in the floor of the cavity using paste epoxy adhesive. Cure under heat and pressure.	Large diameter tear or puncture in honeycomb sandwich panel in *excess of that permitted for* potting compound repair.	Factory fabricated to match ply count, laminate orientation and honeycomb cell size and density. Layup consists only of plug/ patch and adhesive. Easier than separate plug and skin patch, but not practical for highly contoured panels.
Metal Patch Repair	Install a bolted or riveted sheet metal or composite material doubler.	Cracked or punctured structure where load paths, stress levels and safety permit.	Laminate orientation is critical. Due to development of stress concentrations, repair is practical only for secondary fiberglass structure.
Fastener Hole Repair	Apply doubler or fill hole with machineable potting compound or bonded-in-place metal plug. Redrill fastener hole.	Oversized, elongated or mislocated fastener holes where larger fastener cannot be accommodated.	Potting compound or aluminum plug not usable for loaded holes in primary structure. Aluminum specifically not usable with graphite.

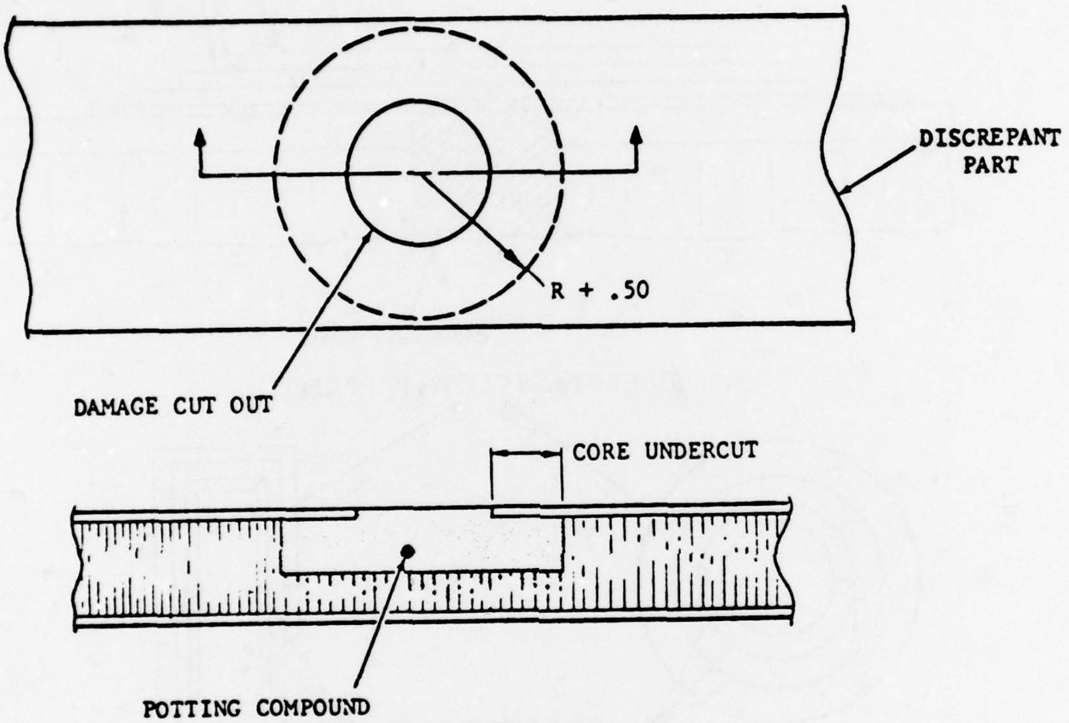
TABLE 38 (Concluded)

Type Repair	Basic Procedure	Damage Description	Comments
Injection Repair	Drill hole pattern through skin over discrepant area. Heat repair area and flow liquid resin into the void. Cure under heat and pressure.	Localized edge damage and delaminations not accompanied by filament fractures.	Used only on service-incurred delaminations. (Fabrication voids usually contain a flaw that prevents rebonding).
Complex Repair	Combination of two or more simple repairs.	Damage to multiple structural elements and/or multiple load paths.	Magnitude and direction of the loads on the structure and the composition of each structural element (plies, orientation, stacking sequence) must be known or determined. Generally impractical for field.
Cannibalization Repair	Cut and remove damaged structure from aircraft. Remove an equivalent piece of structure from a second fuselage and install with structural splices.	Extensive damage to permanent structure.	Custom-engineered repair. Depot level. Requires stocking an entire fuselage or fuselage sections from which replacement parts can be cut.



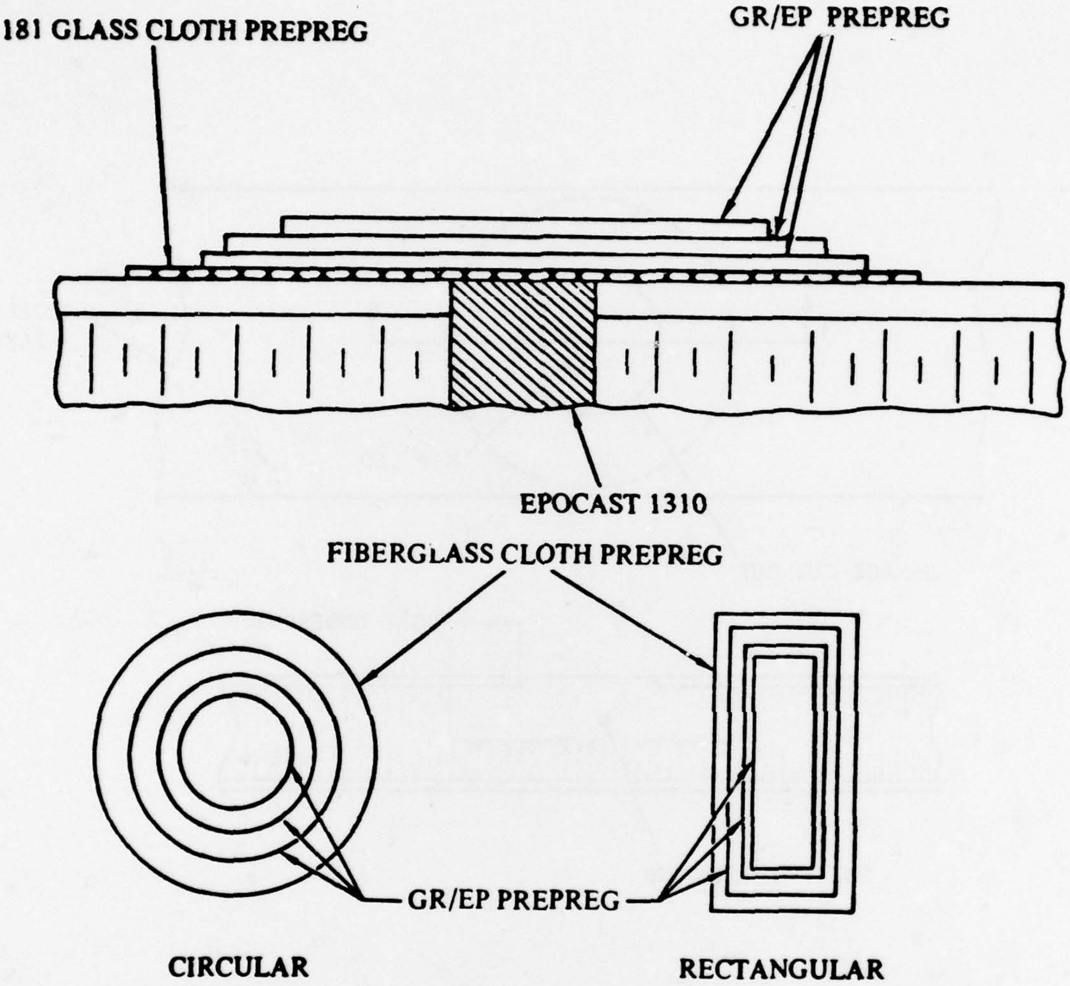
*Reprinted from Reference 15.

Figure 37. Injection Repair



*Reprinted from Reference 16.

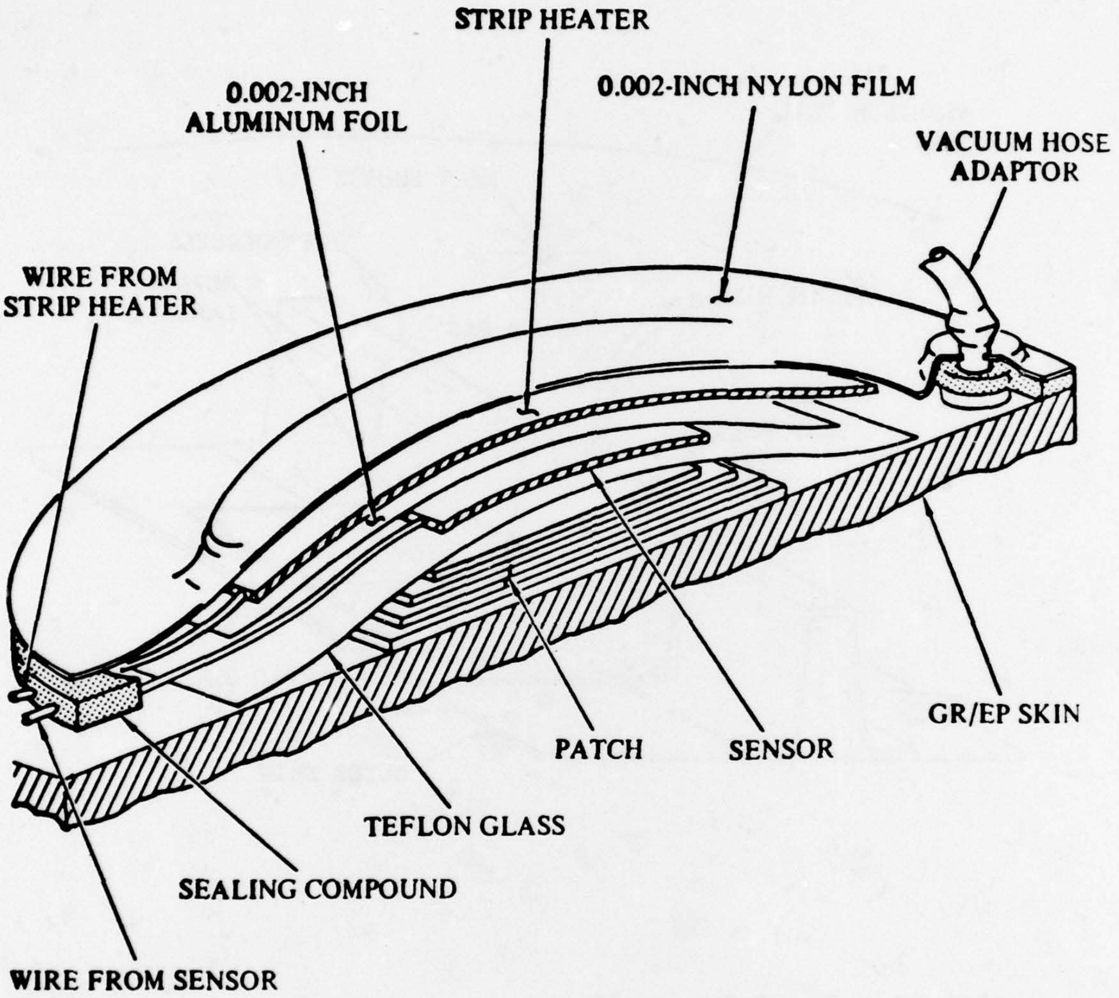
Figure 38. Potting Compound Repair



TOP VIEW

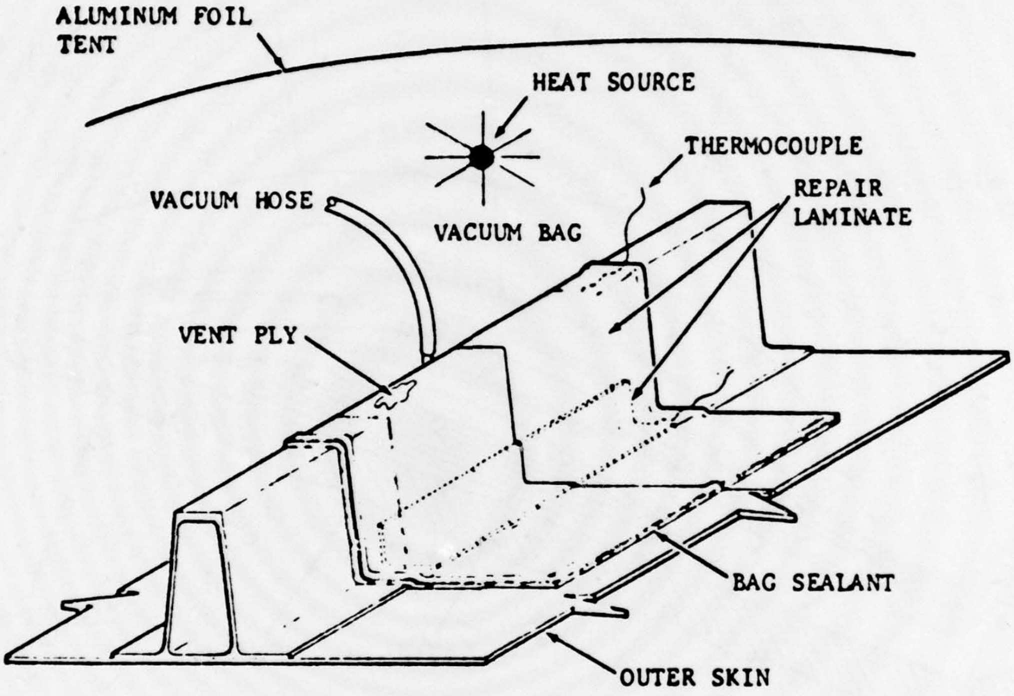
*Reprinted from Reference 15.

Figure 39. Typical Skin Patch Repair



*Reprinted from Reference 15.

Figure 40. Typical Skin Patch Repair Setup



*Reprinted from Reference 16.

Figure 41. Stiffened Sheet Repair

TABLE 29. TYPES OF REPAIR RELATED TO TYPES OF DAMAGE

	Type of Repair											
Type of Damage	Surface Touchup	Surface Refinishing	Injection Repair	Cure-in-Place Skin Patch	Prefabricated Patch/Doubler	Potting Compound Repair	Core Plug Repair	Prefabricated Plug/Patch	Metal Patch Repair	Fastener Hole Repair	Complex Repair	Cannibalization Repair
1. Flaking, peeling, chipping, pitting of surface protection.	X	X										
2. Nicks, scratches, abrasions not damaging laminates.	X	X										
3. Nicks, scratches, abrasions damaging laminate fibers.				X	X							
4. Erosion or fretting of surface material.				X	X							
5. Dents causing delamination and/or core damage.			X	X	X	X	X	X				
6. Cuts or tears in or through sheet or panel.				X	X	X	X	X	X			
7. Punctures or penetrations of structure.				X	X	X	X	X	X			
8. Surface cracks				X	X				X			
9. Subsurface cracks in laminates or core material.				X	X	X	X	X				
10. Delamination of plies or skin-to-core bond.			X	X	X							
11. Crushing, buckling, deformation of structure.											X	X
12. Severing of primary structural member.											X	X
13. Failure of mechanical joint or splice.											X	X
14. Elongated or oversize fastener holes.										X		
15. Fastener holes torn through edge member.				X	X					X		

ADVANCED REPAIR CONCEPTS

Several approaches to repair of advanced composite structures in the field are apparent. For large pieces of structure that are relatively inexpensive to manufacture but difficult to repair extensively in the field, a throwaway concept might be considered. Under this concept, the structure would be designed to be easily replaceable in the field and would be removed and scrapped when major damage was sustained. Tail cones for small to medium sized helicopters are the types of structure that appear to be attractive candidates for this design approach.

For less critical structures, a policy involving more extensive field repair coupled with field expendability in the event of major damage might offer the minimum life-cycle cost solution. Development of low skill level repairs and the use of the new rapid curing adhesives would be emphasized under this approach.

Presenting the most formidable problem are large expensive structures that cannot be repaired in the field when major damage has been suffered. If the structure is field replaceable, the options under present design practices would be to either remove and scrap the structure or return it to depot for repair. If the structure is not field replaceable, as in the case of a cabin roof, the entire airframe would have to be scrapped or returned to depot for repair. Very large repair costs would be suffered in both cases, and in the latter case, extensive time out of service as well.

Combat Damage Repair

In peacetime use of the helicopter, major damage to primary structure of the airframe will occur rarely. For such infrequently occurring events, the cost-effective policy, intuitively, is to return the aircraft for repair at depot rather than incur the logistics and economic penalties of repair in the field. In combat the expected frequency of structural damage increases dramatically, and repair at depot no longer appears to be a viable approach.

Modular Design Approach

The concept of modular design of composite structures has evolved as one of the possible solutions to the problem of combat damage repair. The major concern has to do with damage to large, integral pieces of primary structure that as presently designed and manufactured cannot be easily repaired or replaced in the field. Frames and beams are components of this type.

The concept illustrated in Figure 42 would be to design the structure in sections or modules of a size that can be removed and discarded in the field. Replacement might be accomplished either through the use of mechanical fasteners or through the provision of integral seams along which the structure could be cut. As envisaged these seams would consist of locally reinforced structure which when cut through would provide sufficient

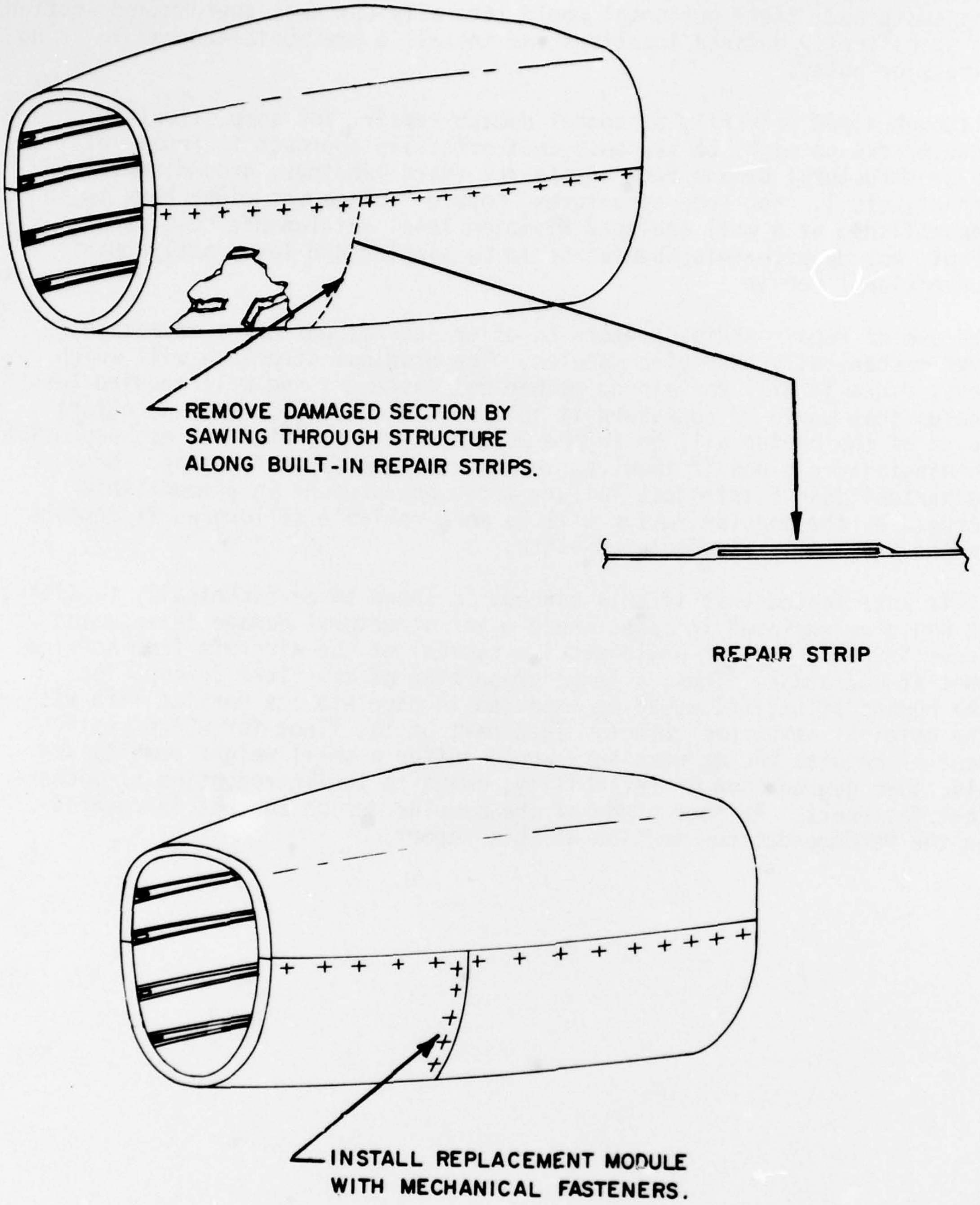


Figure 42. Modular Design Concept

strength for the installation of mechanical fasteners. When heavy damage was sustained, field personnel would literally cut away the damaged section at specifically defined locations and install a prefabricated module using rivets or bolts.

Although aimed primarily at combat damage repair, for some structures modular design might be the most cost effective approach to repair of major structural damage from any source (hard landings, ground vehicle impact, etc.). For some structures, module replacement might have to be accomplished at a well equipped division level maintenance base or at depot, but despite this,modules could be simpler and less costly than conventional repair.

The use of repair strips appears to offer several potential advantages over mechanically assembled modules. The original structure will weigh less, since it will contain no mechanical fasteners and will require less beefup than would be necessary if joints were installed from the outset (part of the beefup will be in the module). It will also be less expensive to manufacture since it involves fewer parts and assembly steps. Because mechanical joints introduce failure modes not present in a monolithic structure, the modular design will be more reliable as long as it remains in the originally manufactured state.

It is anticipated that if this concept is shown to be technically feasible, it would be employed in cases where major structural damage is expected to occur infrequently but would require removal of the aircraft from service when it did occur. Thus, a large proportion of the fleet (except for the combat situation) would be expected to complete its service life with the original structure intact. That part of the fleet for which replacement of modules became necessary would suffer a small weight penalty and also some degradation in reliability, owing to the introduction of mechanical fasteners. Further study of the modular design concept is covered in the Recommendations section of this report.

DAMAGE TOLERANCE AND REPAIRABILITY TESTING

Fuselage skin panels and bulkhead webs typically comprise over 20 percent of helicopter airframe weight. Owing to their exterior location, these areas of the structure are particularly vulnerable to environmental stresses and foreign object damage.

Composite materials provide high strength at low weight for thin fuselage construction. However, resistance to dents and punctures, a major consideration for reliability and maintainability, is also directly related to material thickness. Minimum gauge thicknesses are specified for metallic airframe design, primarily for durability purposes. No such criteria currently exist for composites in any government specifications.

Tests were conducted to assess the damage tolerance and repairability of composite materials typically used in the construction of airframe skin panels and bulkhead webs. The testing covered the commonly used composites of both monolithic and sandwich construction over a range of material thicknesses. The results of the tests were used in part to assess the R&M characteristics of advanced structures concepts and to develop R&M design criteria for these structures.

SCOPE OF TESTING

Monolithic panels were impact tested at varying energy levels to measure the relative damage tolerance of aluminum and three commonly used composite materials. A group of monolithic test specimens was subjected to impact and the damaged specimens were tensile tested to failure to assess the effects of impact damage on structural strength.

A second group of monolithic specimens was damaged by drilling a hole representative of a ballistic penetration through each specimen. The damaged specimens were tensile tested to failure to measure the loss of structural strength produced by this type of damage. A third group of monolithic specimens was damaged in the same manner, the damage was repaired, and the repaired specimens were tensile tested to failure to assess the degree of structural strength restored by simple field-type repairs.

Sandwich panels employing combinations of composite and aluminum facing materials and aluminum and Nomex honeycomb core were impact tested at varying energy levels to measure the relative damage tolerance of these types of construction.

A group of sandwich panel test specimens was subjected to impact and the damaged specimens were beam flexure tested to failure to assess the effects of impact damage on structural strength. A second group of sandwich panel test specimens was subjected to impact at the same energy level. The resulting damage was repaired and the repaired specimens were beam flexure tested to failure to assess the degree of structural strength restored by simple field-type repairs.

TEST METHODS

Impact Testing of Monolithic Panels

A total of 64 monolithic panels was fabricated for impact testing. An equal number of panels (16 each) were fabricated from fiberglass/epoxy, Kevlar/epoxy, graphite/epoxy and aluminum. Table 40 describes the material, thickness, ply orientation and stacking sequence of the panels, each of which was made approximately 6 inches square.

TABLE 40. MONOLITHIC IMPACT TEST SPECIMENS

Material	Qty.	Thickness (inch)	Laminate Layup
7781/5143	4	.020	(0, 90)
10 mil	4	.040	(0, 90) 2
Fiberglass/	4	.060	(0, 90) 3
Epoxy	4	.080	(0, 90) 4
AS/RAC 6350	4	.024	(0, 90, 0)
8 mil	4	.040	(0, 90, 0, 90, 0)
Graphite/	4	.056	(0, 90, 0, 90, 0, 90, 0)
Epoxy	4	.072	(0, 90, 0, 90, 0, 90, 0, 90)
285/5143	4	.020	(0, 90)
10 mil	4	.040	(0, 90) 2
Kevlar/	4	.060	(0, 90) 3
Epoxy	4	.080	(0, 90) 4
2024-T3	4	.016	-
Aluminum	4	.025	-
Alloy	4	.032	-
	4	.040	-
Total	64		

The impact tests were performed with a dart impact tester (Figure 43). Each specimen was clamped to a rigid metal frame and placed on a hollow square metal base with the center of the panel aligned with the vertical cylinder containing the impact projectile, a 2-pound, 0.75-inch-diameter, spherical-nosed weight. The projectile, guided within the vertical cylinder, was dropped from various heights corresponding to impact energies of 20, 30, 40 and 50 inch-pounds, one of each set of four panels impacted once at one of the four energy levels. The energy levels were chosen to represent the type of impact that would be caused by dropping typical hand tools. After each impact test, the type and size of the resulting damage were recorded.

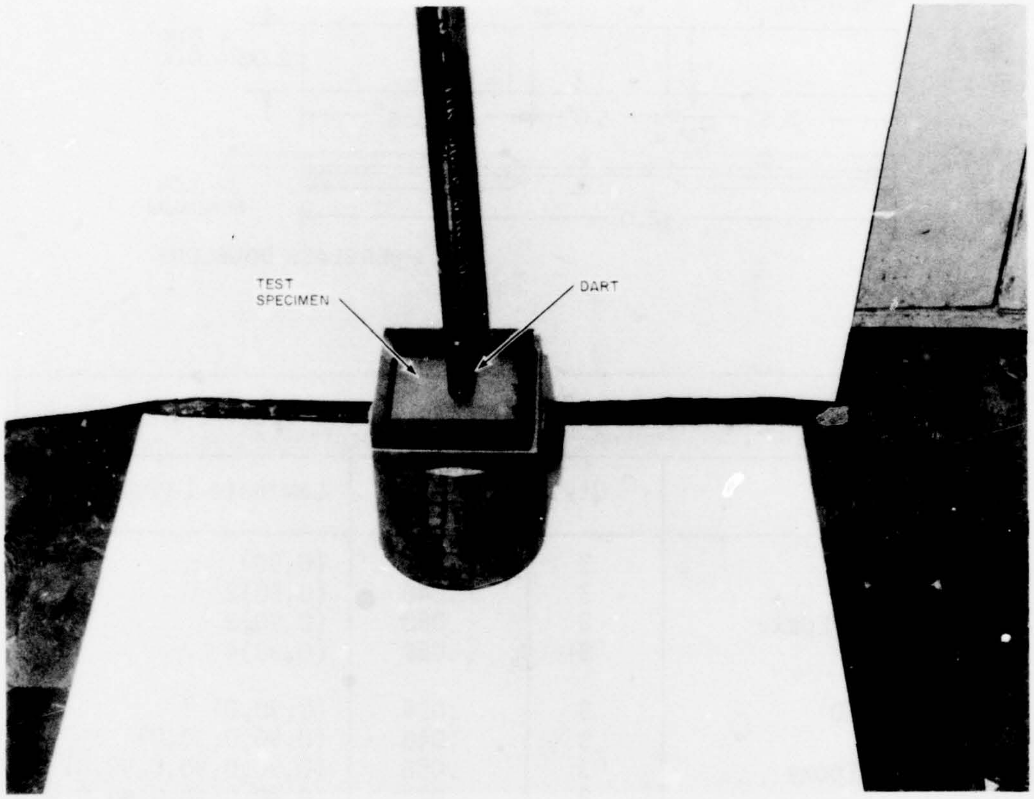


Figure 43. Impact Test Setup

Impact and Tensile Testing of Monolithic Panels

A total of 48 monolithic test specimens was fabricated for impact and tensile testing. An equal number of specimens (12 each) were fabricated from fiberglass/epoxy, Kevlar/epoxy, graphite/epoxy and aluminum. Table 41 lists the materials, thicknesses, ply orientation and stacking sequence of the test specimens. The configuration of the metallic and nonmetallic specimens is also shown.

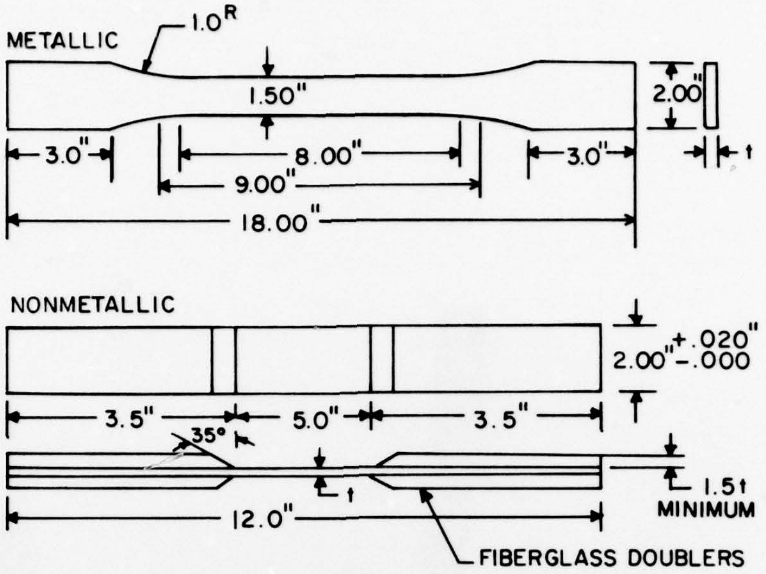


TABLE 41. MONOLITHIC IMPACT AND TENSILE TEST SPECIMENS

Material	Qty.	Thickness (t)(inch)	Laminate Layup
7781/5143	3	.020	(0,90)
10 mil	3	.040	(0,90)2
Fiberglass/Epoxy	3	.060	(0,90)3
	3	.080	(0,90)4
AS/RAC 6350	3	.024	(0,90,0)
8 mil	3	.040	(0,90,0,90,0)
Graphite/Epoxy	3	.056	(0,90,0,90,0,90,0)
	3	.072	(0,90,0,90,0,90,0,90)
285/5143	3	.020	(0,90)
10 mil	3	.040	(0,90)2
Kevlar/Epoxy	3	.060	(0,90)3
	3	.080	(0,90)4
2024-T3	3	.016	
Aluminum	3	.025	
Alloy	3	.032	
	3	.040	
Total	48		

Two of each set of three test specimen configurations were impacted at an energy level of 60 inch-pounds using the dart impact tester previously described. After each impact the type and size of the resulting damage were recorded. The undamaged specimen and one of the two damaged specimens of each configuration were then tensile tested to failure in a Riehle 20,000-pound-capacity FA-20 testing machine at a cross-head speed of .20 inch per minute (Figure 44). The average load level at which the damaged specimens failed was compared to the load level at which the undamaged specimen failed to measure the loss of strength produced by the impact damage.

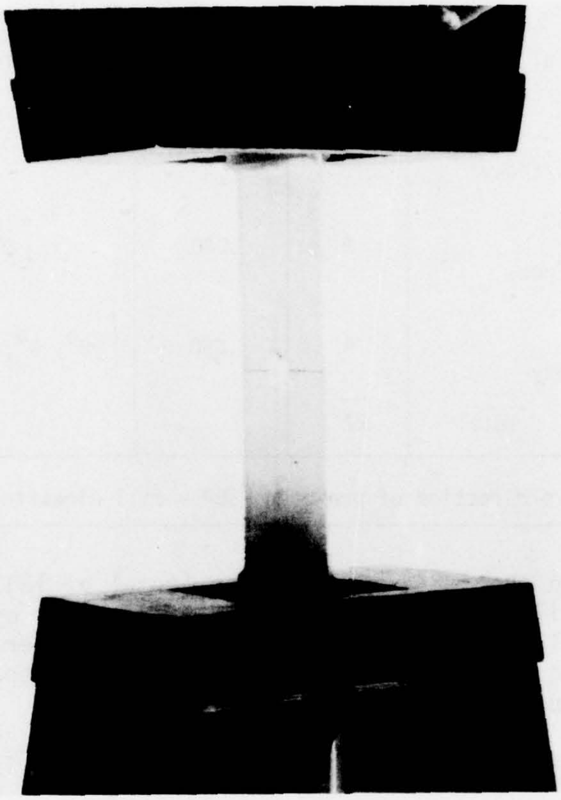


Figure 44. Tensile Test Setup

Originally it was planned to repair the second of the two damaged specimens of each configuration and tensile test it to failure to assess the effectiveness of the repair. This was not done for reasons explained later in the discussion of the test results.

Through-Damage Repair and Tensile Testing of Monolithic Panels

A total of 27 monolithic test specimens was fabricated for through-damage repair and tensile testing. An equal number (9 each) were fabricated from fiberglass/epoxy, Kevlar/epoxy and graphite/epoxy. The configuration of the test specimens was as shown in the sketch accompanying Table 41. The material, thickness, ply orientation and stacking sequence of the 27 specimens are given in Table 42.

TABLE 42. MONOLITHIC THROUGH-DAMAGE REPAIR AND TENSILE TEST SPECIMENS

Material	Qty.	Thickness (inch)	Laminate Layup*
7781/5143 10 mil Fiberglass/Epoxy	9	.040	$[0^\circ]_4$
AS/RAC 6350 8 mil Graphite/Epoxy	9	.040	$[0^\circ]_4$
285/5143 10 mil Kevlar Epoxy	9	.050	$[90^\circ, 0^\circ, 90^\circ, 0^\circ, 90^\circ]$
Total	27		

* 0° = warp direction of pre-preg; 90° = fill direction.

In six of the nine specimens of each type (total of 18), a 5/16-inch-diameter hole was drilled through the approximate center of the gage section to represent a ballistic penetration of 7.62 mm caliber. Half of each group of specimens containing the drilled hole were repaired using simple field-type procedures.

Monolithic Graphite Repair

The graphite specimens were repaired with titanium sheet as follows: Loose splinters surrounding the drilled hole were removed. Patches were cut from .016-inch-thick annealed titanium sheet (MIL-T-9046, Type III, Composition C-6AL-4V) as shown in Figure 45. The bonding surface of each patch was abraded with fine sandpaper and the surfaces of the patches and specimens were cleaned with solvent. EA9309.2 paste adhesive (Hysol Division, Dexter Corp.) was applied to the surfaces of the specimen to be repaired using scrim cloth for uniform thickness. A titanium patch was applied to one side of the specimen and pressure was applied with a plate and clamps using a parting film over the patch. The patch was allowed to cure at room temperature for 24 hours and a second patch was applied to the opposite side of the specimen using the same procedure.

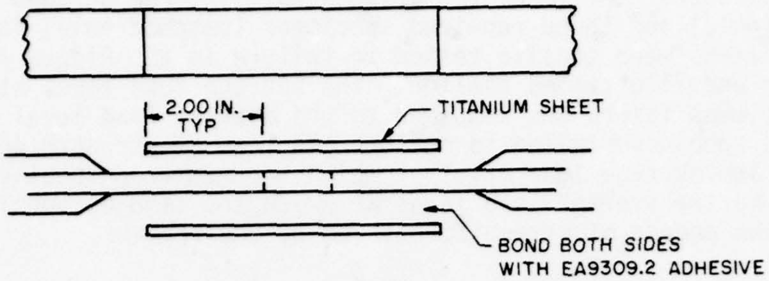


Figure 45. Monolithic Graphite Repair

Monolithic Fiberglass and Kevlar Repair

Two circular patches were cut from Type 181 fiberglass cloth as shown in Figure 46. The surface of the test specimen was cleaned with solvent and the two-ply patch was applied to one side using a mixture of Epon 828 resin and 10% catalyst Type DTA. A pressure caul separated from the patch with a parting cloth was used to apply pressure, and the patch was allowed to cure at room temperature for 24 hours. The cavity formed by the drilled hole in the specimen closed on one side by the patch was filled with EA9309.2 paste adhesive and allowed to cure. A two-ply fiberglass patch was then applied to the opposite side of the specimen in the same manner as the first.

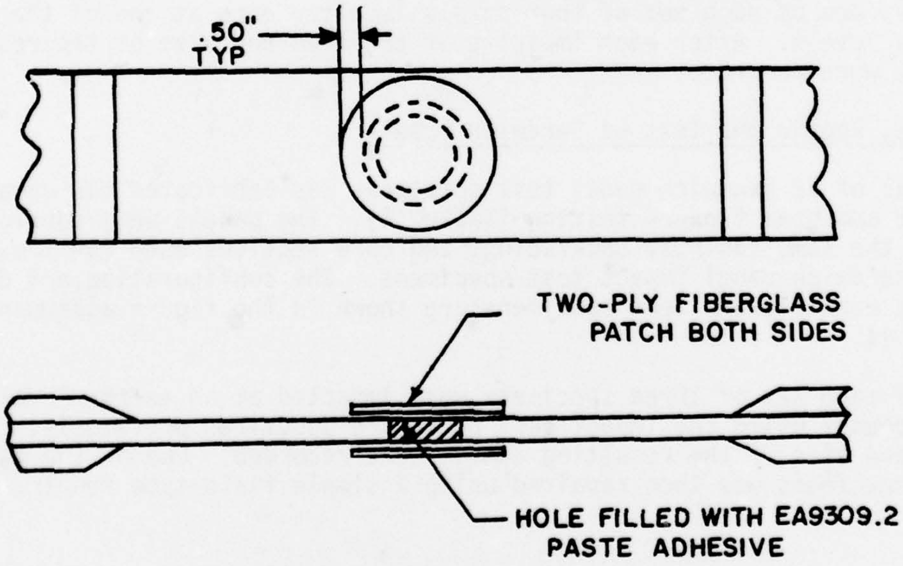


Figure 46. Monolithic Fiberglass and Kevlar Repair

The three control specimens (no drilled hole), three damaged specimens (drilled hole) and three repaired specimens (patched hole) in each set of nine specimens were tensile tested to failure in the Riehle testing machine described and illustrated earlier. The average load level at which the damaged specimens failed was compared to the average load level at which the undamaged specimens failed to measure the loss of strength caused by the damage. The average load level at which the repaired specimens failed was compared to the average load level at which the damaged specimens failed to measure the degree of strength restored by the repair.

Impact Testing of Sandwich Panels

A total of 96 sandwich panels was fabricated for impact testing. An equal number of panels (24 each) were fabricated with facings of fiberglass/epoxy, Kevlar/epoxy, graphite/epoxy and aluminum. The panels were fabricated with facings of the same materials, thicknesses, ply orientation and stacking sequence used for the equivalent monolithic panels listed in Table 40. Backfacings were made .020 inch thick for the nonmetallic panels and .016 inch thick for the metallic panels. Half of each specimen group (12 each) were fabricated with Nomex honeycomb core and half with aluminum honeycomb core. Both core materials had a 3/16-inch cell size and a density of 3 pounds per cubic foot. Each panel was a minimum of 6 inches square. The sandwich panel impact test specimens are listed in Table 43.

The impact testing was conducted with the same test setup used to conduct the monolithic panel impact tests. The sandwich panels were centered on the square metal base and were impacted by the 2-pound projectile dropped from distances corresponding to impact energies of 20, 30, 40 and 50 inch-pounds, one of each set of four panels impacted once at one of the four energy levels. After each impact test the type and size of the resulting damage were recorded.

Damage, Repair and Test of Sandwich Panels

A total of 72 sandwich panel test specimens was fabricated for damage, repair and beam flexure testing (Table 44). The panels were fabricated using the same facings, backfacings and core sections used for the equivalent sandwich panel impact test specimens. The configuration and dimensions of the beam flexure test specimens are shown in the figure accompanying Table 44.

Two of each set of three specimens were impacted at an energy level of 60 inch-pounds using the impact test procedure described previously. The type and size of the resulting damage were recorded. One of the two damaged specimens was then repaired using a simple field-type repair.

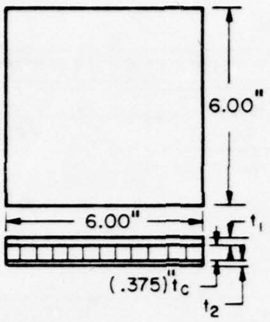
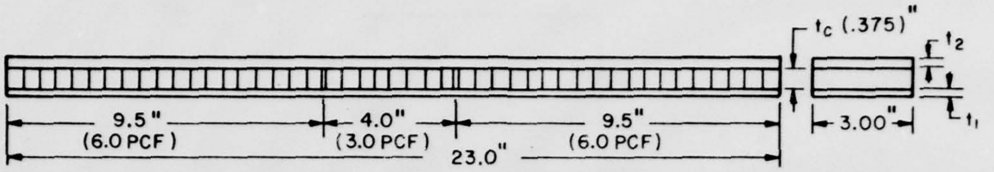


TABLE 43. SANDWICH PANEL IMPACT TEST SPECIMENS

Facing Material	Sandwich Core				Qty.	Thickness (inch)	
		Size (inch)		Density			
	Mat'l	Cell	Foil	lb/ft^3		T$_1$	T$_2$
7781/5143	HRH 10	3/16	.002	3.0	4	.020	.020
10 mil	Nomex		.005	6.0	4	.030	.020
					4	.040	.020
Fiberglass/	5052	3/16	.001	3.1	4	.020	.020
Epoxy	Alum.		.002	5.7	4	.030	.020
					4	.040	.020
AS/RAC 6350	HRH 10	3/16	.002	3.0	4	.024	.024
8 mil	Nomex		.005	6.0	4	.032	.024
					4	.040	.024
Graphite/	5052	3/16	.001	3.1	4	.024	.024
Epoxy	Alum.		.002	5.7	4	.032	.024
					4	.040	.024
285/5143	HRH 10	3/16	.002	3.0	4	.020	.020
10 mil	Nomex		.005	6.0	4	.030	.020
					4	.040	.020
Kevlar/	5052	3/16	.001	3.1	4	.020	.020
Epoxy	Alum.		.002	5.7	4	.030	.020
					4	.040	.020
2024-T3	HRH 10	3/16	.002	3.0	4	.016	.016
Aluminum	Nomex		.005	6.0	4	.020	.016
Alloy					4	.032	.016
	5052	3/16	.001	3.1	4	.016	.016
	Alum.		.002	5.7	4	.020	.016
					4	.032	.016
				Total	96		



PCF = lb/ft^3

TABLE 44. SANDWICH PANEL IMPACT AND REPAIR TEST SPECIMENS

Facing Material	Sandwich Core				Qty.	Thickness(in)	
	Mat'l	Size (inch)		Density lb/ft^3		T_1	T_2
		Cell	Foil				
7781/5143 10 mil Fiberglass/Epoxy	HRH 10 Nomex	3/16	.002 .005	3.0 6.0	3 3 3	.020 .030 .040	.020 .020 .020
	5052 Alum.	3/16	.001 .002	3.1 5.7	3 3 3	.020 .030 .040	.020 .020 .020
AS/RAC 6350 8 mil Graphite/Epoxy	HRH 10 Nomex	3/16	.002 .005	3.0 6.0	3 3 3	.024 .032 .040	.024 .024 .024
	5052 Alum.	3/16	.001 .002	3.1 5.7	3 3 3	.024 .032 .040	.024 .024 .024
285/5143 10 mil Kevlar/Epoxy	HRH 10 Nomex	3/16	.002 .005	3.0 6.0	3 3 3	.020 .030 .040	.020 .020 .020
	5052 Alum.	3/16	.001 .002	3.1 5.7	3 3 3	.020 .030 .040	.020 .020 .020
2024-T3 Aluminum Alloy	HRH 10 Nomex	3/16	.002 .005	3.0 6.0	3 3 3	.016 .020 .032	.016 .016 .016
	5052 Alum.	3/16	.001 .002	3.1 5.7	3 3 3	.016 .020 .032	.016 .016 .016
				Total	72		

Graphite-Faced Sandwich Panel Repair

The damaged facing and core material was removed from the panel, leaving any large splinters extending outside the area of extensive damage to be bonded in place during the repair. The cutout core area was filled with syntactic foam, allowed to cure and sanded flush with the facing. Patches were cut from .016-inch-thick annealed titanium sheet (MIL-T-9046, Type III, Composition C-6AL-4V) as shown in Figure 47. The bonding surfaces of the titanium patches were lightly abraded and these surfaces and the surface of the panel were thoroughly cleaned with solvent.

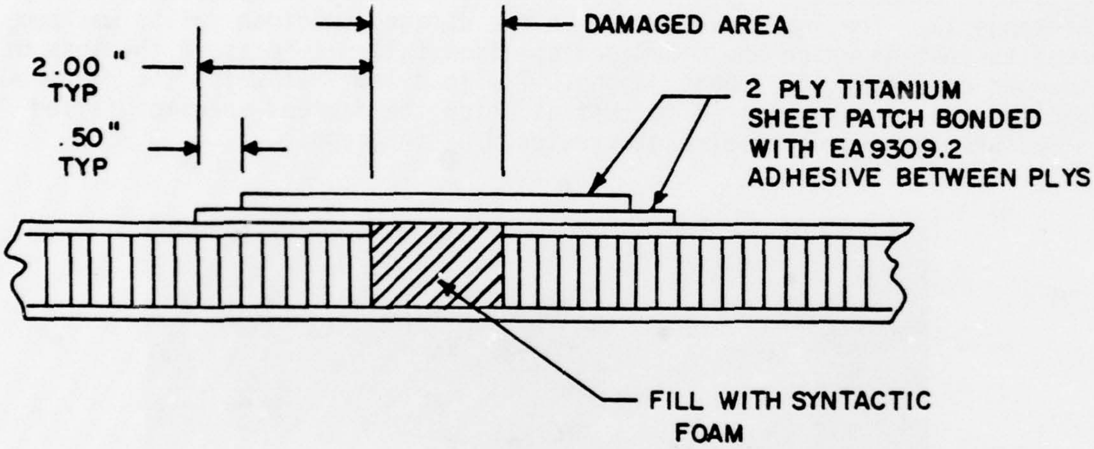


Figure 47. Graphite-Faced Sandwich Panel Repair

The larger of the two patches was bonded to the surface of the panel over the damaged area using EA 9309.2 paste adhesive and scrim cloth to provide uniform bond thickness. A parting film was placed over the patch and weighted to apply pressure. The patch was allowed to cure for 24 hours at room temperature, following which the second smaller diameter patch was bonded to the first using the same procedure.

Repair of Fiberglass and Kevlar-Faced Sandwich Panels

Damaged facing and core material was removed and the damaged core area was filled with syntactic foam, allowed to cure and sanded flush with the facing. A two-ply fiberglass patch was applied in the manner described for repair of the fiberglass and Kevlar monolithic test specimens.

Repair of Aluminum-Faced Sandwich Panels

Repair of the aluminum-faced sandwich panels was essentially the same as the repair of the graphite-faced panels except that aluminum sheet was used in lieu of titanium sheet to form the patches.

The control specimen, damaged specimen and repaired specimen in each set of three specimens were beam flexure tested to failure. Testing was conducted in a Riehle 20,000-pound-capacity FS-20 testing machine (Figure 48). The tests were conducted using a two-point loading method in accordance with Reference 17. The load level at which the damaged specimen failed was compared to that at which the undamaged specimen failed to measure the loss of strength caused by the impact damage. The load level at which the repaired specimen failed was compared to that at which the damaged specimen failed to measure the degree of strength restored by the repair.

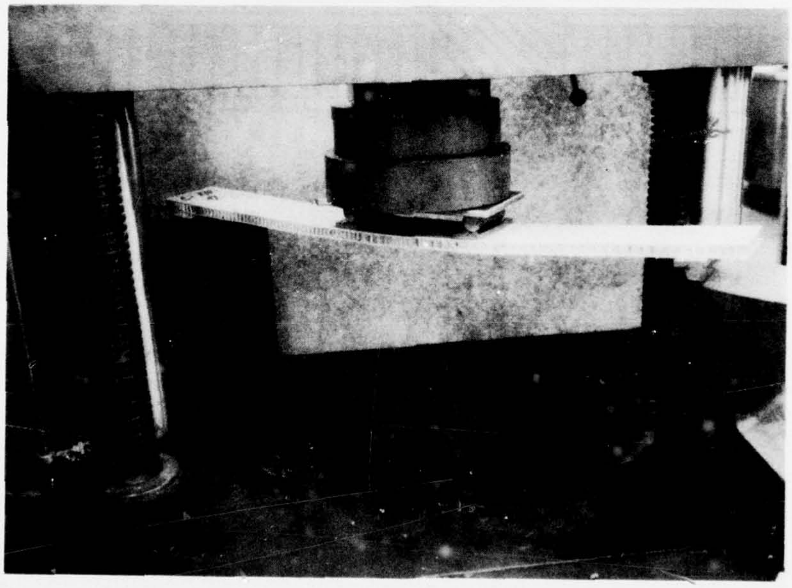


Figure 48. Beam Flexure Test Setup

---

[17] American Standard Test Method ASTM C393-62, Flexure Test of Flat Sandwich Constructions (Reapproved 1970).

TEST RESULTS

The results of the damage tolerance and repairability testing are described next. Detailed test results are presented in Appendix B.

Impact Tests of Monolithic Panels

Figure 49 summarizes the results of the monolithic panel impact testing. For impact energies corresponding to typical hand tool drops (20-30 inch-lb range), all three of the composites and the aluminum appear to show acceptable damage tolerance, based on the visible damage sustained. Typical damage is shown in Figures 50 through 54. A thickness of .020 inch appears to represent a minimum gage for composites in applications where minor impact is expected at some significant frequency.

For higher impact energy levels the thin composites show a tendency to fracture, whereas the aluminum, because of its capacity to yield, tends to dent. In the case of the thinnest Kevlar and graphite panels, complete penetration occurred at the 50-inch-lb energy level. This suggests that the minimum gage for composites should be increased to .040 inch or thicker in applications where frequent impact at higher energy levels (dropped parts, shifting cargo, etc.) is anticipated.

Figure 55 shows the general behavior of aluminum and monolithic composites subjected to impact. Aluminum is characterized by progressively deeper denting as the impact energy level increases. Because of their inability to yield under stress, composites typically experience three stages of damage. At low energy levels the composites either experience no damage or suffer minor subsurface damage (local delamination) appearing as a local blemish or discoloration of the laminate. A point is reached at which impact begins to produce visible damage in the form of broken fibers and surface fractures. These appear both on the side of the impact and on the opposite side of the laminate and become progressively more severe until complete penetration of the material occurs. The unidirectional graphite experienced substantial splintering of the opposite face at the higher impact energy levels. The energy levels at which the three types of damage occur are dependent on such factors as the thickness and configuration of the laminate and the shape of the impacting object.

For equivalent thicknesses, the amount of damage sustained at low energy levels indicates the following ranking of damage tolerance:

Fiberglass (best)

Graphite (second best)

Kevlar (poorest)

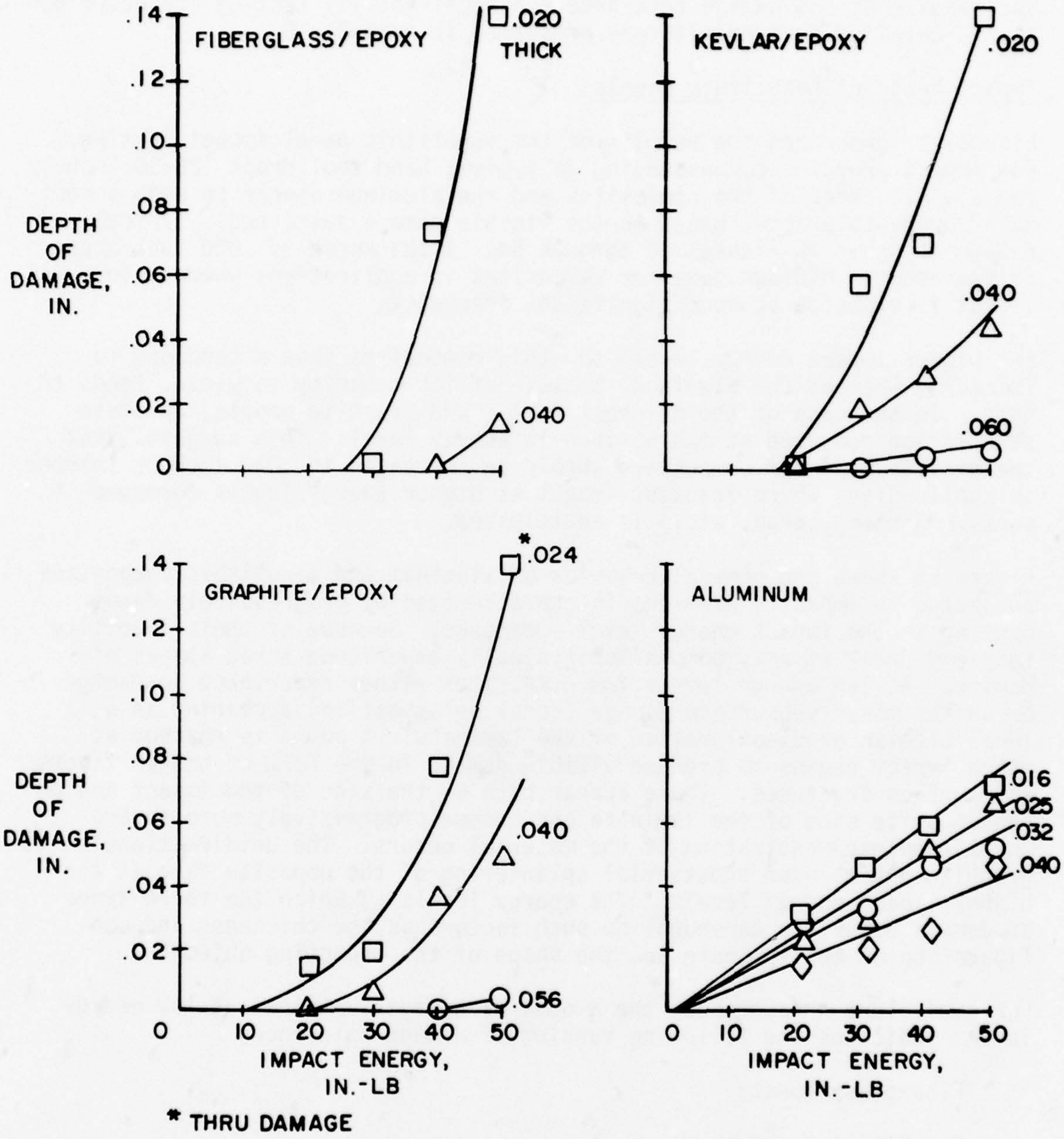


Figure 49. Summary of Monolithic Panel Impact Testing

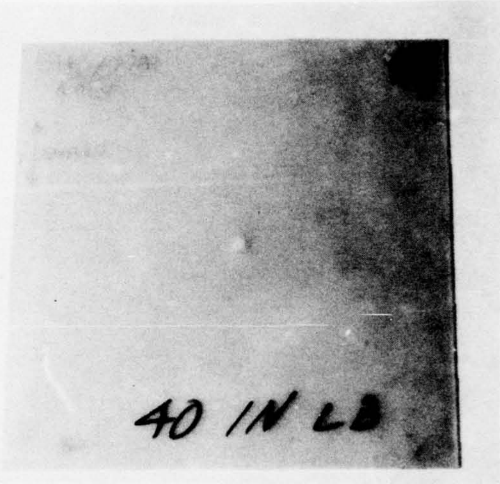


Figure 50. Typical Subsurface Damage

Figure 51. Typical Fracture

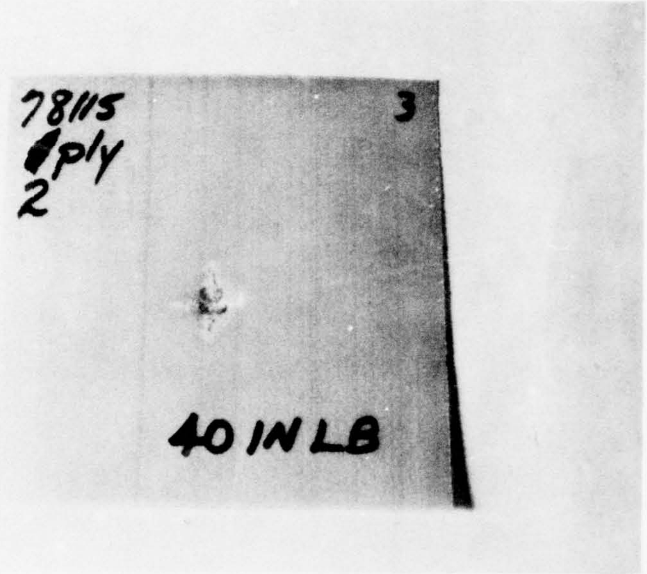


Figure 52. Typical Penetration

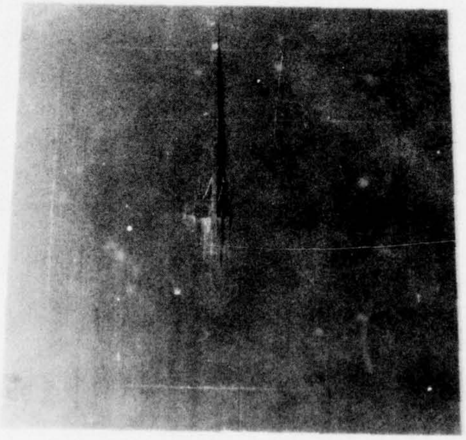


Figure 53. Typical Splintering of Unidirectional Graphite Panel

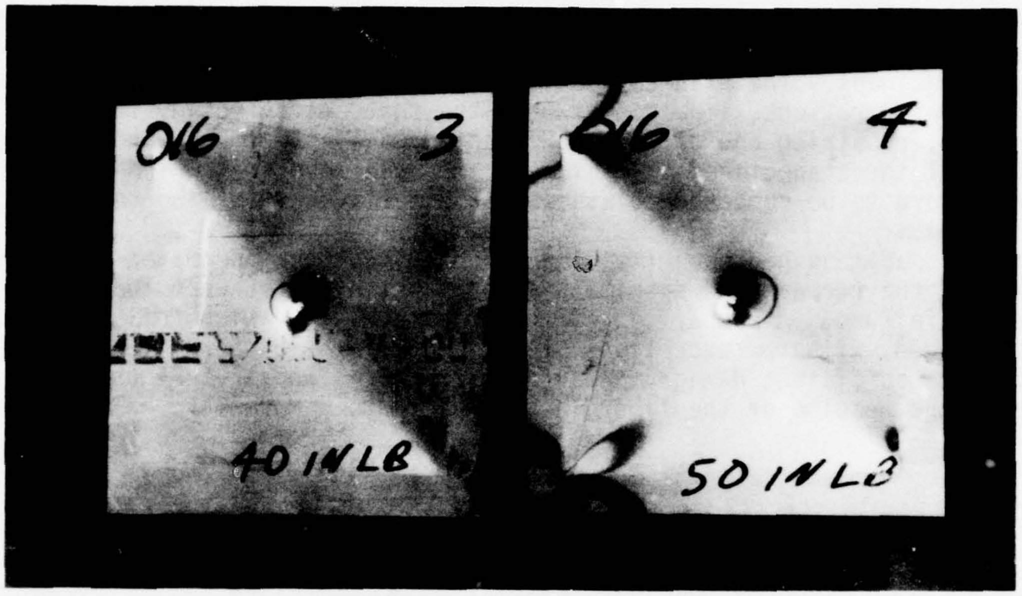


Figure 54. Typical Impact Damage to Aluminum Panels

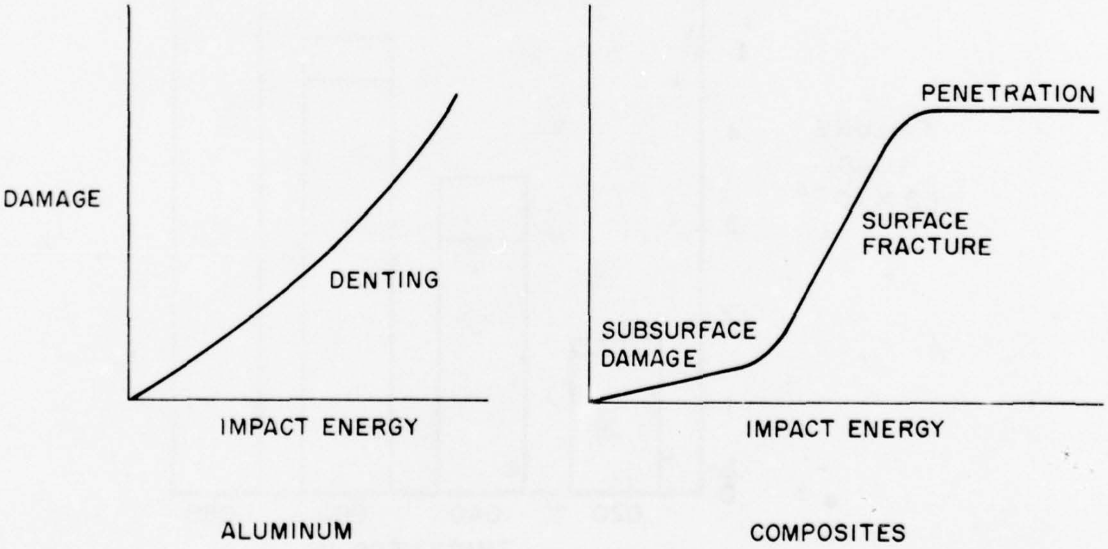


Figure 55. Characteristic Behavior of Aluminum and Composites Subjected to Impact

Impact and Tensile Tests of Monolithic Panels

Originally it was planned to investigate the relative repairability of the three composite materials and aluminum by repairing and testing impact-damaged specimens. The plan called for building three specimens each of several configurations, impacting two of the three at an energy level of 60 inch-lb, repairing one of the two impact-damaged specimens, and tensile testing all three specimens to failure. The load levels at which failure occurred were to be compared to assess the effectiveness of the repair.

After the specimens had been impacted as described, the undamaged specimen and one of the two damaged specimens in each set of three were tensile tested to failure. A comparison of the load levels at which failure occurred showed, with few exceptions, that the 60-inch-lb energy level did not produce sufficient damage to warrant structural repair. Figure 56 presents the results of the fiberglass panel tests. Moreover, calculated

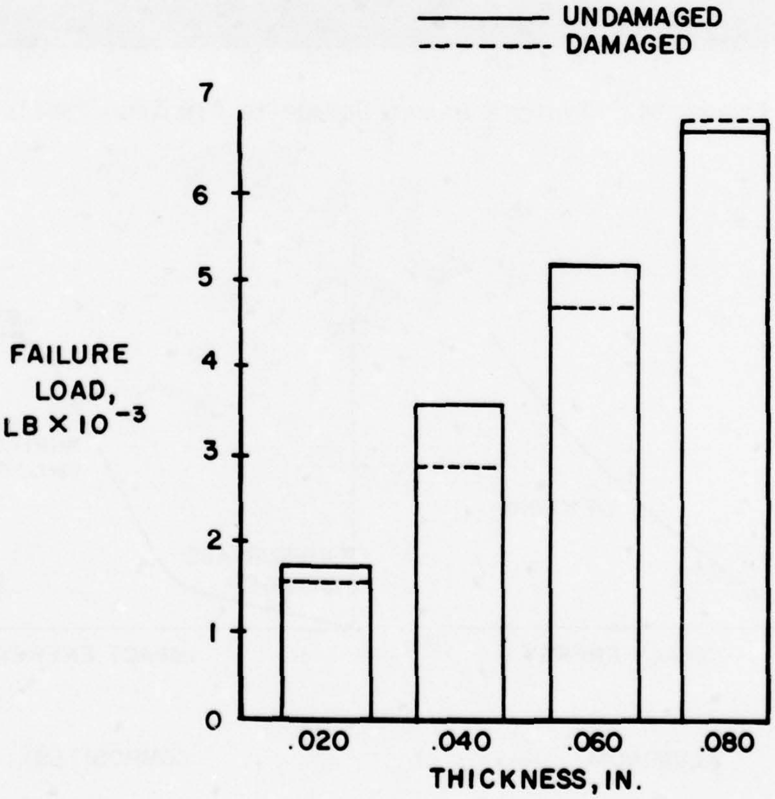


Figure 56. Summary of Tensile Tests of Fiberglass Monolithic Panels Damaged by 60-Inch-Lb Impact

failure stresses showed a number of anomalies with published material properties data. These were believed to be due to normal statistical scatter, but since the plan called for a single test of each configuration, there was no way to reconcile them. It was concluded that it would not be productive to conduct the repair phase of the testing, and a more positive test involving multiple test samples was proposed to and accepted by the Army.

Through-Damage Repair and Tensile Testing of Monolithic Panels

Under the revised plan the repairability of the three monolithic composite materials was assessed by repairing and testing specimens damaged by simulated ballistic penetrations. Test procedures and repair methods were described earlier. Repair of aluminum was not included in the monolithic panel testing because field repair methods for aluminum were already well established. Calculations were made to compare analytically the strength in tension of a repaired aluminum sheet configured like the composite test specimens. Figure 57 shows a typical riveted repair on which the calculations were based.

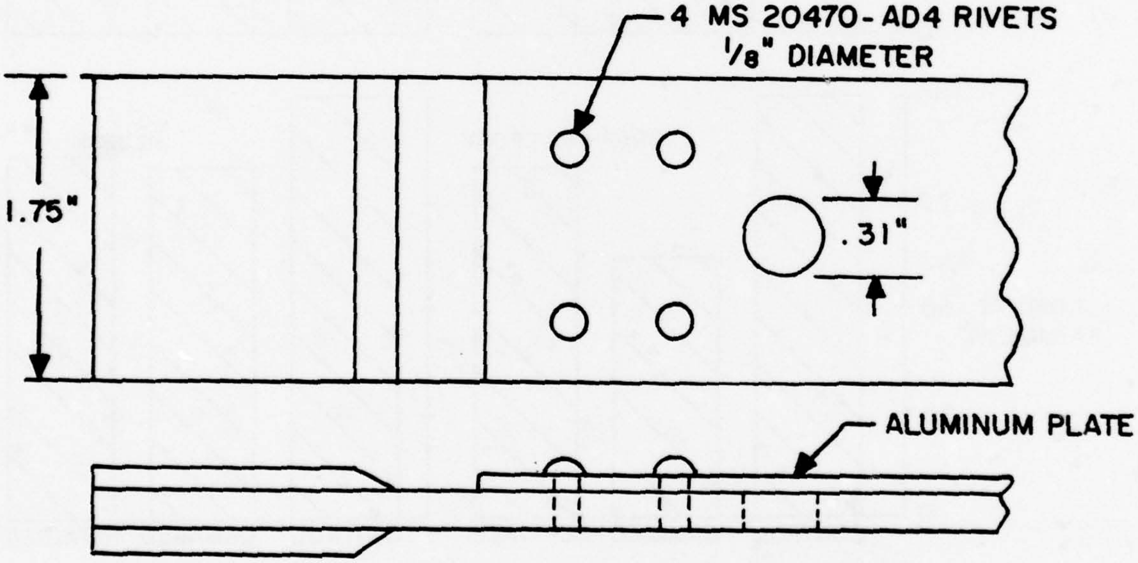


Figure 57. Repair of Aluminum Tensile Test Specimen Damaged via a Drilled Hole

Figure 58 shows the results of these tests. Also shown are the comparable values calculated for aluminum. In each case the 5/16-inch drilled hole simulating a ballistic penetration caused a significant loss of tensile strength. Reductions in load capability averaged approximately 50 percent for the fiberglass/epoxy and the Kevlar/epoxy and approximately 40 percent for the graphite/epoxy. Typical failures are shown in Figure 59.

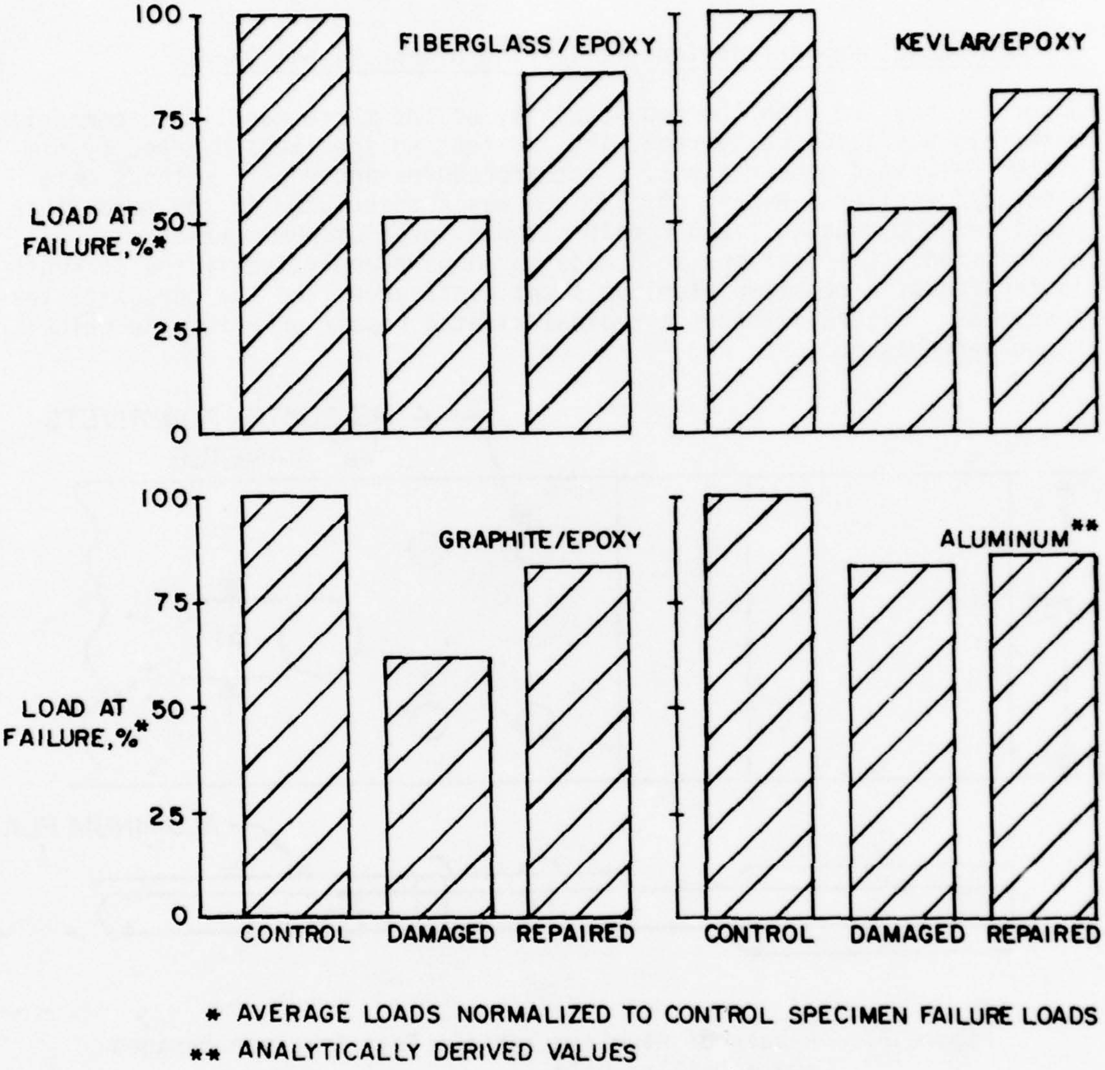


Figure 58. Summary of Tensile Testing of Through-Damaged and Repaired Monolithic Panels

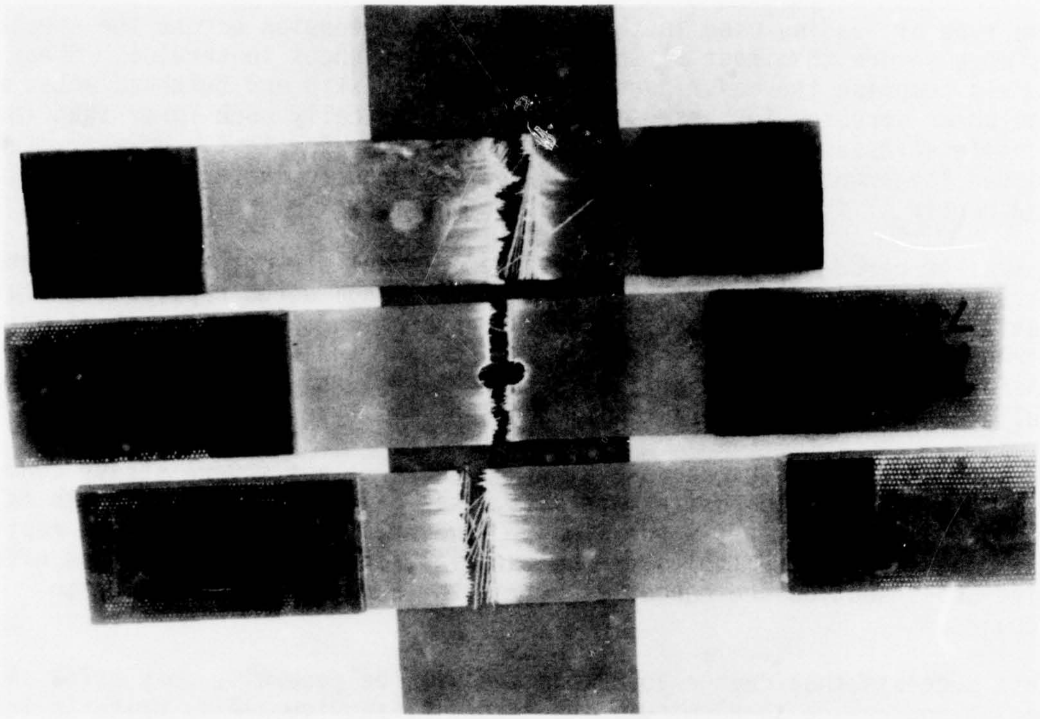


Figure 59. Typical Failures of Monolithic Test Specimens

A substantial restoration of strength was achieved with the field-type repairs. Increases in load capability averaged approximately 65 percent for the fiberglass/epoxy, 50 percent for the Kevlar/epoxy and 35 percent for the graphite/epoxy.

In every case, failure of the repaired specimens occurred outside the area of the drilled hole. With the fiberglass and Kevlar specimens, failure occurred at the edge of or slightly beyond the patch. Failure of the graphite specimens occurred initially as a separation of the bond between the titanium patch and the graphite, followed by a failure through the drilled hole. In all cases the repair was successful in reducing the stress concentration at the hole. However, other stress concentrations created by the repairs themselves became the points of failure.

The simple field-type repairs employed in this test succeeded in restoring the specimens to within 80 to 85 percent of their original strength. If the load level failing the control specimens is viewed as an ultimate load, the repairs were successful in restoring the strength of the specimens to a value comfortably above limit load (typically 2/3 of ultimate). In practice, airframe structures should never be subjected to ultimate loads in service.

The type of loading used in this test (uniform tension across the specimen) is more severe than most of the airframe experiences in service. Shear panels comprise the major part of the aircraft skin and bulkhead webs, and the shear stresses in these structures are typically much lower than the tensile stresses applied to the specimens during the test. However, tension loaded longerons and beam caps could be designed for high tensile loads, and repair of these components might be considered a potential problem area.

A smaller percentage reduction from the original strength could undoubtedly have been achieved through the use of custom-engineered repairs (carefully built up and tapered patches, etc.). Restoration to 100 percent of original strength is probably impossible in cases where the structural element is uniformly loaded in tension (as in these tests), since no matter how carefully engineered, the repair will develop some type of stress concentration.

As shown in Figure 58, because aluminum suffers less severe stress concentrations than the composite materials, less of its original strength is lost when equivalent damage is sustained. However, the typical riveted repair introduces additional holes in the material, and thus restores less effective cross section and hence less strength than the bonded composite repairs.

This suggests that damage to composites will be generally more critical than equivalent damage to aluminum. As a result, serviceability criteria for composites will have to be more specifically defined, particularly that related to deferrability of damage. Further development work in the area of quick-fix field repairs is also required.

Sandwich Panel Impact Test Results

Figures 60 and 61 present the results of the sandwich panel impact tests. As shown by the plotted data, for all four facing material configurations, the aluminum honeycomb panels sustain a greater degree of measurable indentation than the equivalent Nomex honeycomb panels. Damage to the composite-faced panels of both core types included shallow dents and either fractures or complete penetrations of the facing material. Damage to the composite-faced aluminum honeycomb panels tended toward deeper dents and fewer fractures, while that of the composite-faced Nomex honeycomb panels tended toward fewer and shallower dents and more frequent fractures. The aluminum-faced sandwich panels of both core types dented more readily than the composite-faced panels but did not fracture.

The aluminum honeycomb panels suffered a greater degree of measurable damage than the Nomex panels, and also had a greater propensity for denting versus fracture because the aluminum honeycomb tends to crush upon impact and remain depressed, whereas the Nomex tends to break or crack upon impact and then return to its original shape. Figure 62 illustrates the two types of core damage. As a result the Nomex honeycomb panels tend to sustain less surface damage upon impact but also to suffer more hidden subsurface damage. During the tests, some of the Nomex panels showing minor surface damage after impact were discovered to have detectable subsurface damage

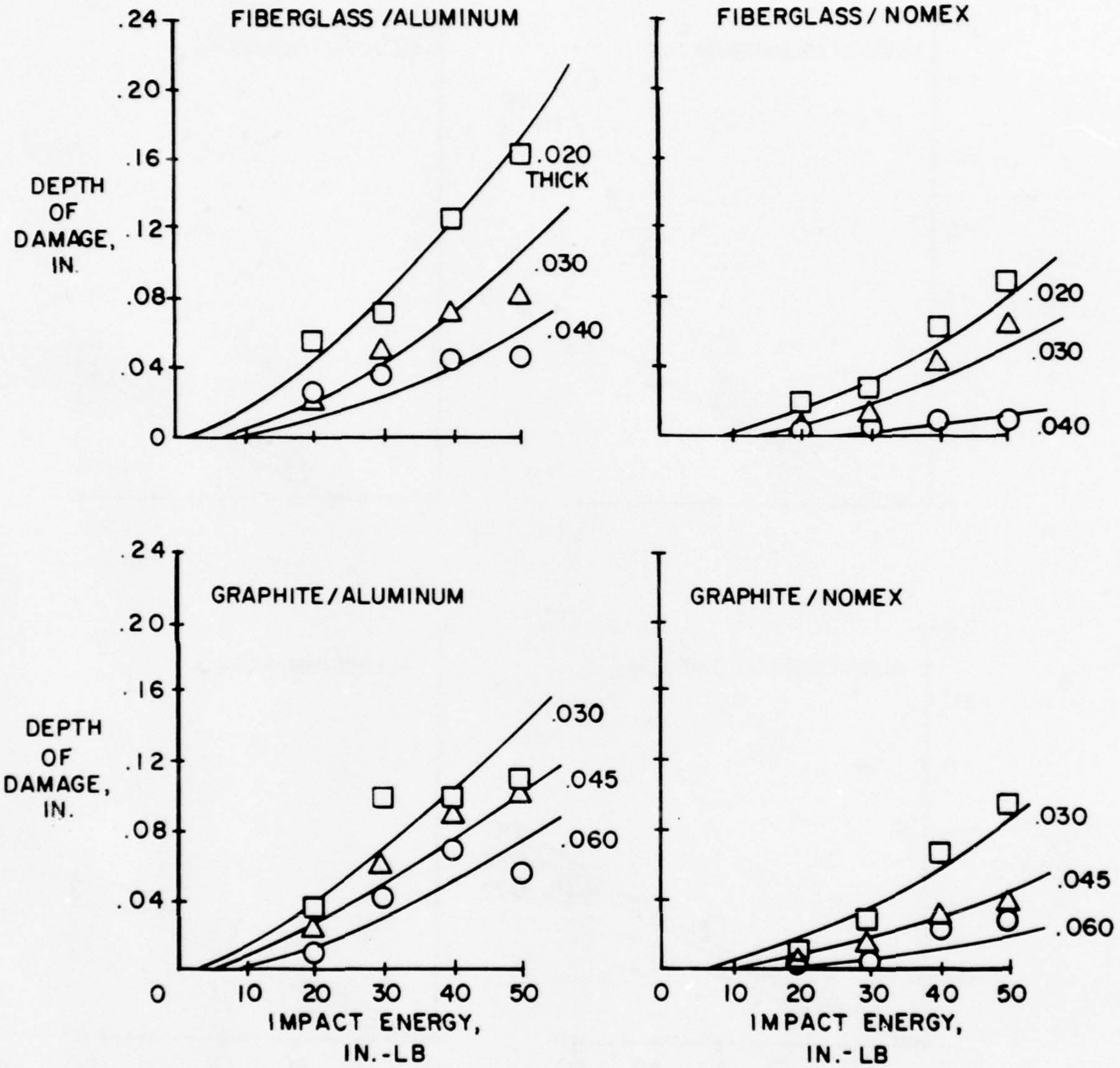


Figure 60. Summary of Fiberglass and Graphite-Faced Sandwich Panel Impact Testing

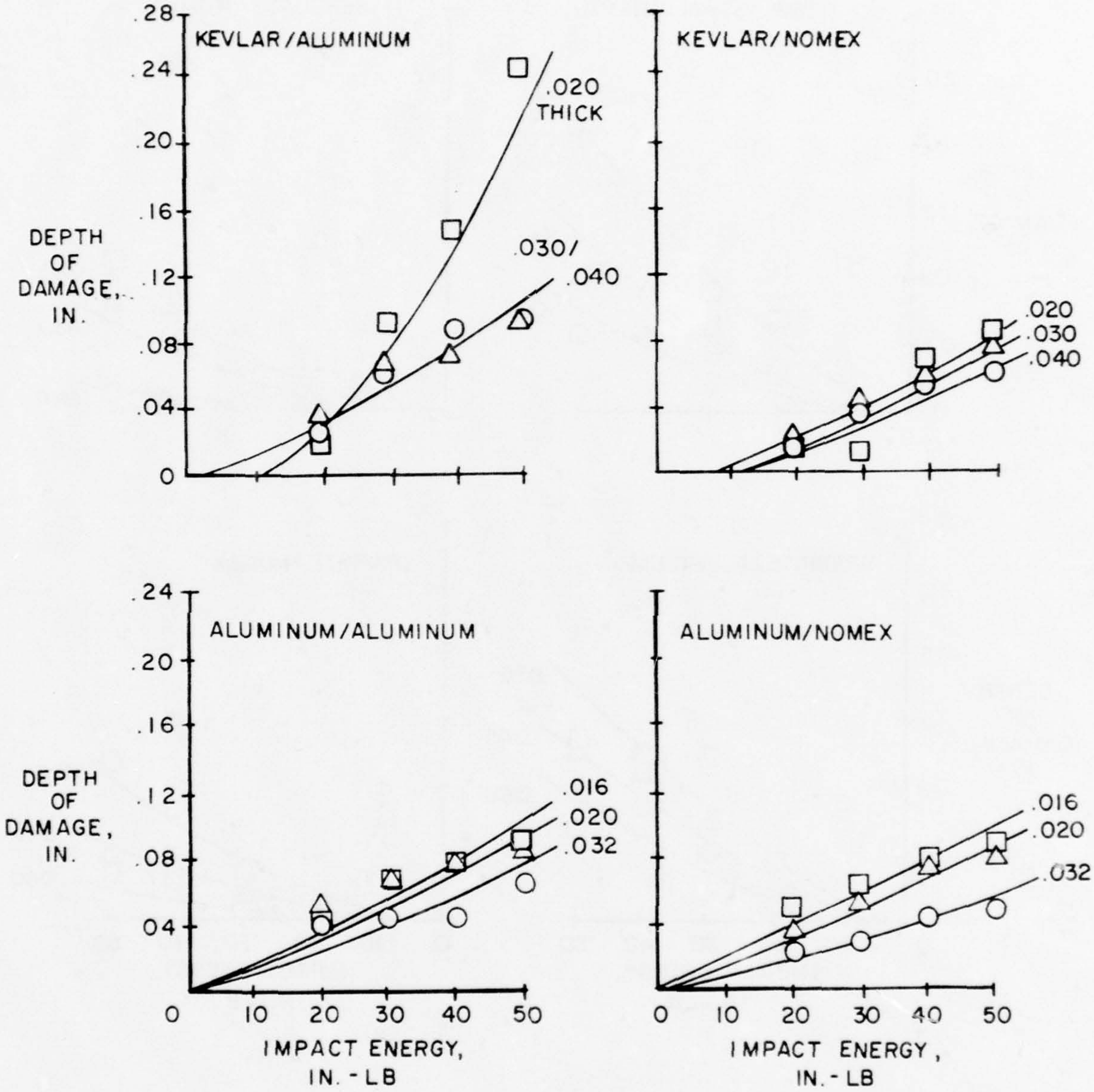


Figure 61. Summary of Kevlar and Aluminum-Faced Sandwich Panel Impact Testing

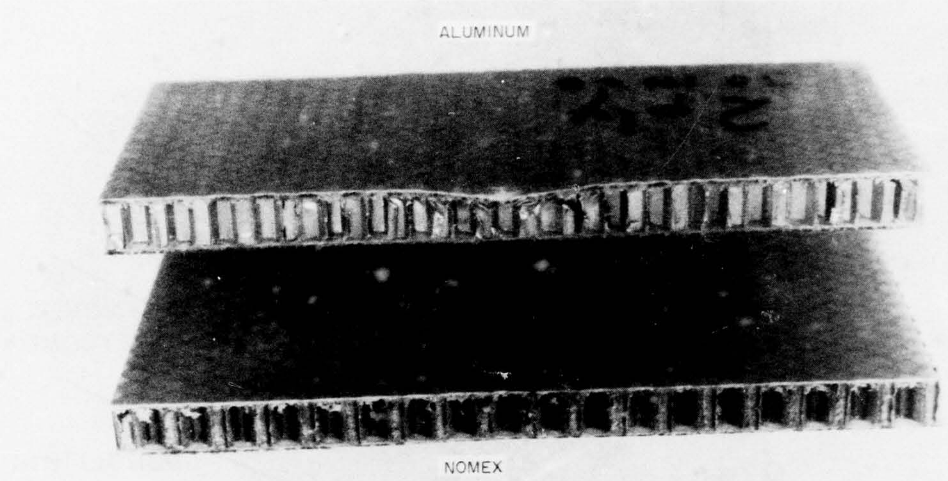


Figure 62. Typical Core Damage Sustained by Sandwich Panels Subjected to Impact

evidenced by an ability to locally depress the facing and a crinkling sound of the damaged core beneath.

A rating of the impact tolerance of the sandwich panels, based on visible surface damage, shows the Nomex core panels to be superior in every case. For the Nomex panels the amount of damage sustained appears to be only moderately affected by the thickness of the facing and largely independent of the facing material, whether composite or aluminum. Damage to the aluminum honeycomb panels appears to be much more affected by the thickness of the facing. The aluminum-faced/aluminum honeycomb panels appear to be more damage tolerant than the composite-faced/aluminum honeycomb panels, among which no significant variation in damage tolerance is apparent.

For all three of the composite materials, and to a lesser extent for the aluminum, the material tends to suffer greater damage when used as the facing of a sandwich panel than it does in monolithic form. A comparison of damage versus impact energy is shown for 0.040-inch-thick Kevlar in Figure 63. The reduced damage tolerance of materials used in sandwich panel facings is due to the greater stiffness provided by the sandwich form. In monolithic form the composites are resilient and tend to resume their original shape after moderate impact. When used as a facing of a sandwich panel, the materials have less flexibility and thus must absorb more energy.

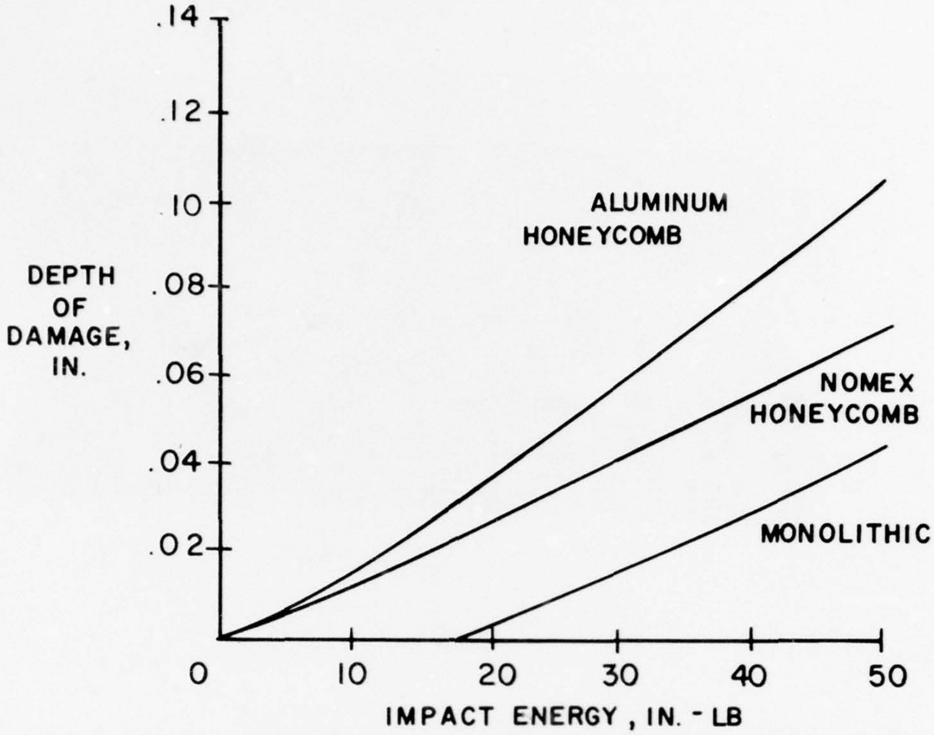


Figure 63. Relative Damage Tolerance of .040-Inch-Thick Kevlar Used as a Sandwich Panel Facing and in Monolithic Form

Also, the core material when crushed (especially the aluminum) tends to stay permanently deformed and prevent the facing from resuming its flat shape unless the bond is broken.

Sandwich Panel Damage, Repair and Beam Shear Test Results

Figures 64 and 65 present the results of the sandwich panel damage, repair and beam shear tests. Unlike the monolithic panel impact testing reported on earlier, impact at an energy level of 60 inch-pounds did cause a significant loss of strength in the sandwich panels. All tests were conducted with the damaged face on the compression side of the panels, and all panels experienced buckling failures through the damaged area, indicative of a loss of compression stability. A typical failure is shown in Figure 66. Significant from the standpoint of R&M is the fact that the Nomex panels, while exhibiting significantly less surface damage, appear to suffer a loss of strength due to impact roughly equivalent to that of the aluminum honeycomb panels. This probably would not be true for the tension side of the panel, however, where the integrity of the facing would provide the primary resistance to failure. The fact that damage can be sustained without visible evidence may present field inspection problems for some types of structure.

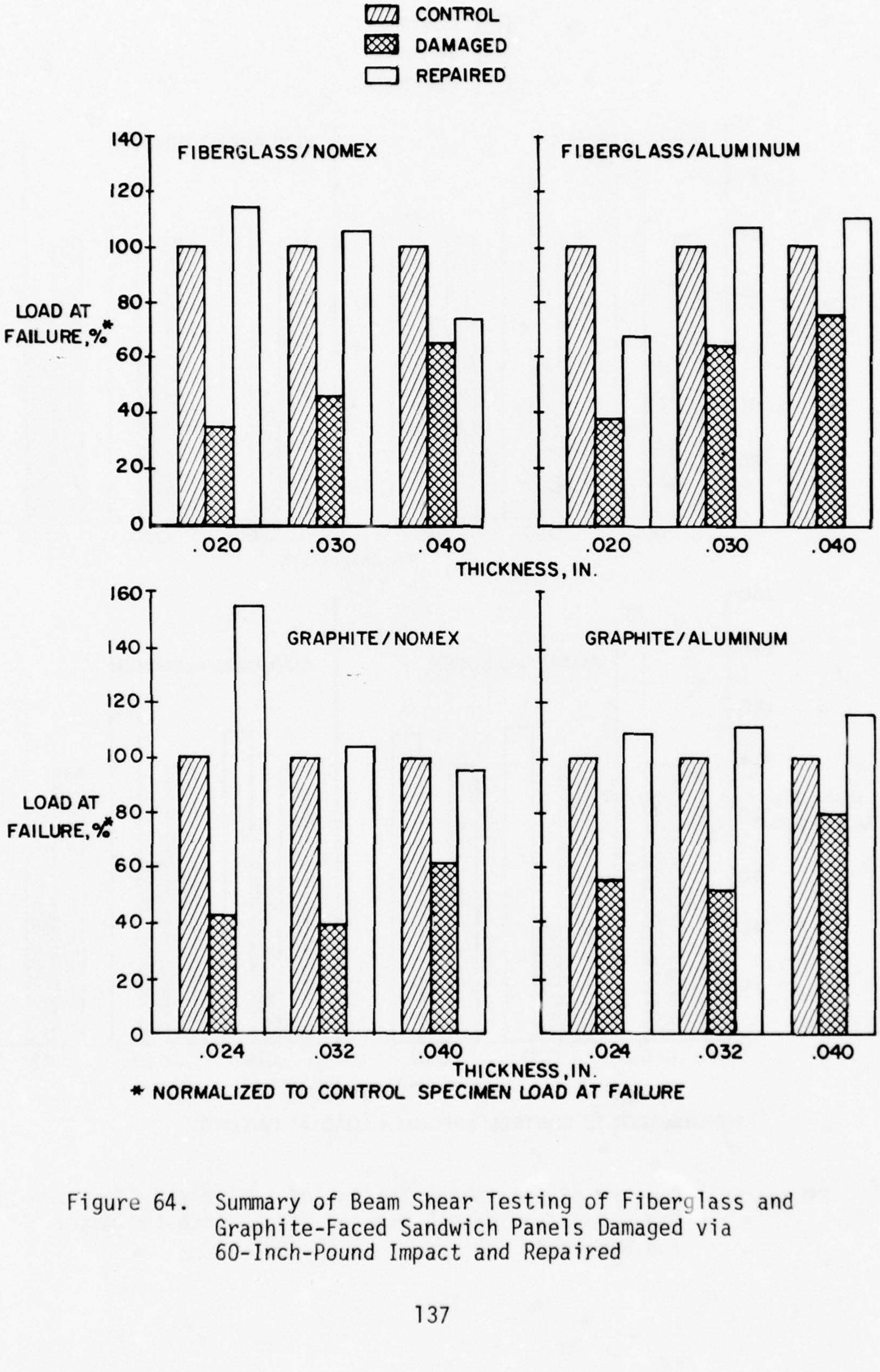


Figure 64. Summary of Beam Shear Testing of Fiberglass and Graphite-Faced Sandwich Panels Damaged via 60-Inch-Pound Impact and Repaired

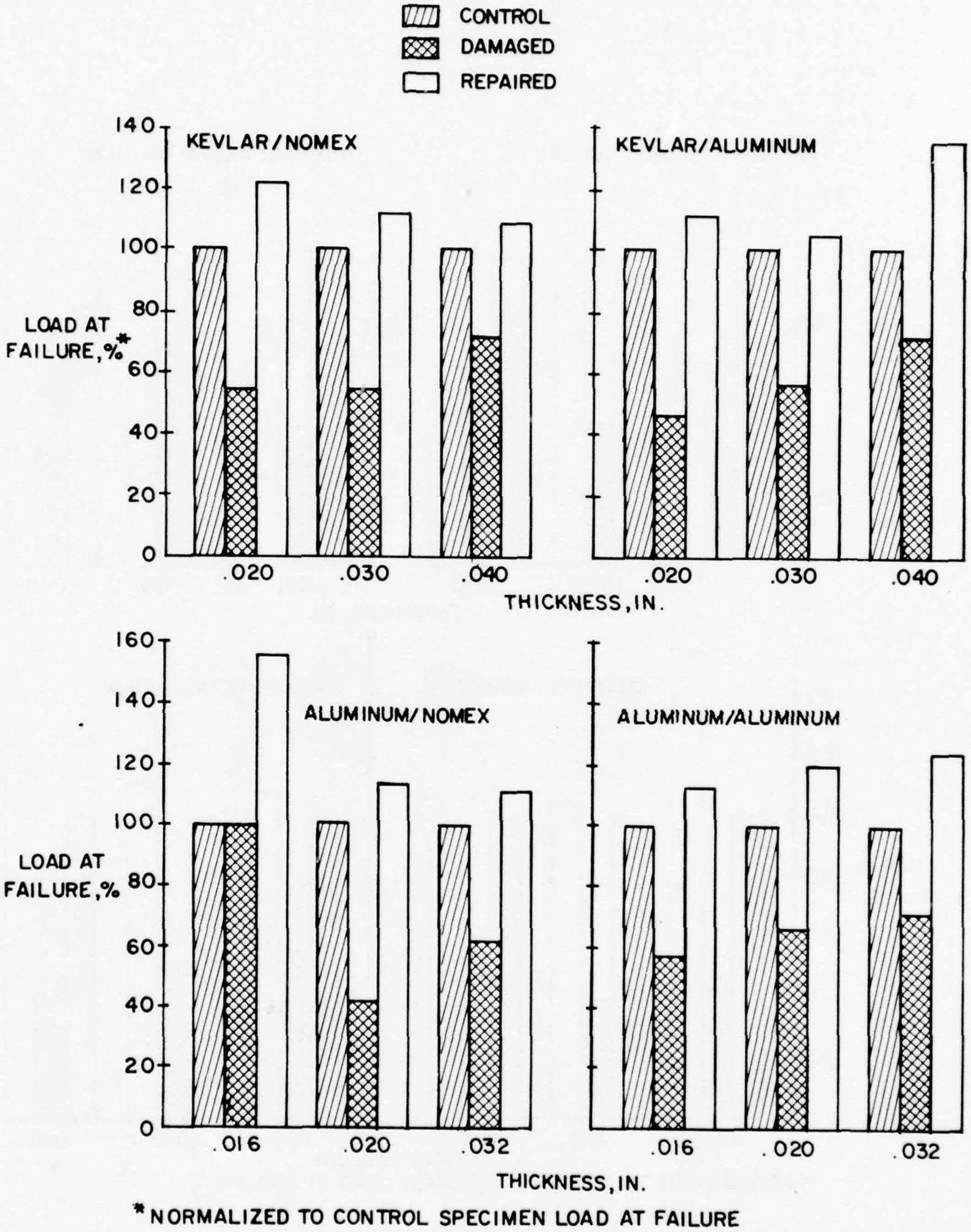


Figure 65. Summary of Beam Shear Testing of Kevlar and Aluminum-Faced Sandwich Panels Damaged Via 60-Inch-Lb Impact and Repaired

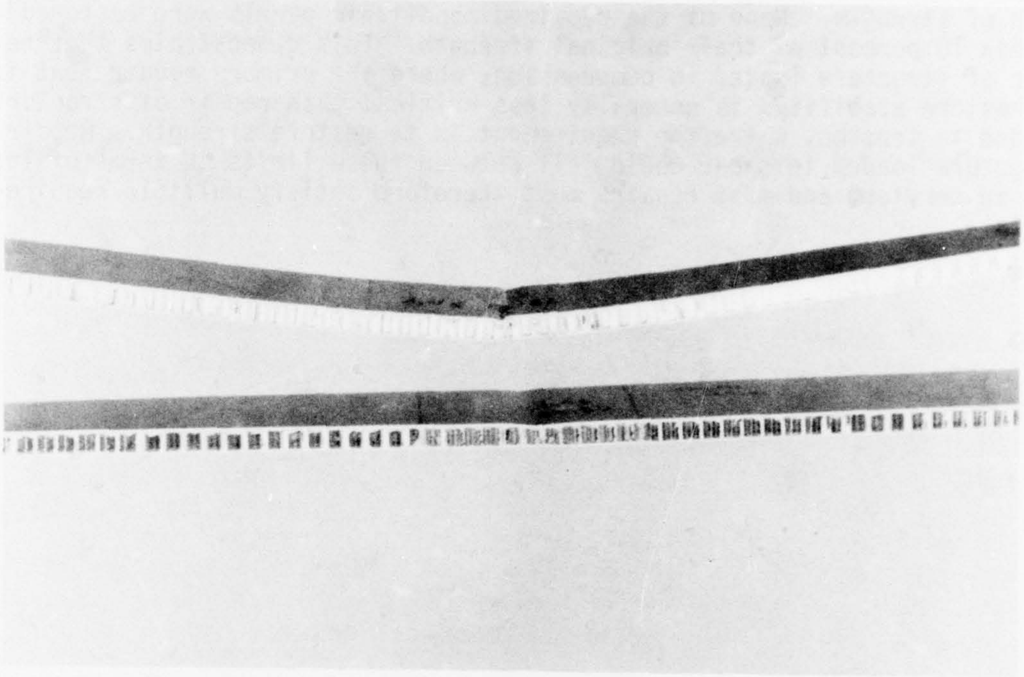


Figure 66. Typical Failure of Sandwich Panel Beam Shear Specimen

As shown by the plotted test data, the field-type methods used to repair the damaged sandwich panels succeeded (with just two exceptions) in restoring most or all of the strength to the panel. In a number of cases the repaired panels actually failed at a higher load than the undamaged panels. The effectiveness of the repairs is attributed to the added stiffness they provided to the panel, which in turn improved compression stability and prevented buckling within the repaired area. Most failures of repaired panels occurred as buckling of the panel at the edge of repair.

Two of the repaired fiberglass-faced panels, one with Nomex core and one with aluminum core, failed at substantially lower load levels than the respective control specimens. This may have been caused by random variation in the repair procedures or test methods, or possibly by the presence of undetected core damage extending outside the area of the patch. The design of sandwich panel repairs should consider this possibility and, where significant, specify a larger patch than might be indicated by the size of the visible damage alone. This may be particularly significant for the Nomex core panels which tend to suffer more hidden damage.

As reported earlier, equivalent types of repair applied to the monolithic panels that were tested in tension did not provide an equivalent restoration of strength. None of the repaired monolithic panels were restored to within 10 percent of their original strength. This demonstrates that repair of structure loaded in compression, where the primary requirement is to restore stability, is generally less critical than repair of structure loaded in tension, where the requirement is to restore strength. Repair of structure loaded in shear would fall between these limits in terms of loading in service, and most repairs must therefore satisfy multiple requirements.

R&M/COST ASSESSMENT TECHNIQUE

One of the objectives of this program was to develop an R&M and cost assessment techniqe for advanced structures concepts. The initial approach to developing a technique was quantitative, based on a system of numerical weights and scores which were used to assess the various characteristics of a design. Attempts to apply the technique to actual designs did not produce satisfactory results, however, and after several modifications a basically qualitative approach evolved. The difficulties that were encountered with quantitative assessment are reviewed briefly before describing the final technique.

ORIGINAL QUANTITATIVE APPROACH

Because of the lack of experience data on which to base numerical R&M predictions, the weighting and scoring values used with the original method were chosen to represent relative rankings and order-of-magnitude differences in design attributes suggested by engineering judgment and analysis. Table 45 summarizes the original technique.

In that scheme damage potential was one of the variables evaluated via the method of numerical weighting and scoring. Damage potential, it was reasoned, is related to an aircraft's exposure to environmental hazards and to the level of exposure of specific components of the airframe to these hazards. Since both of these factors can vary widely based on the type of aircraft, its mission and operating environment, there are no quantitative values that can be used to express then universally. A simple weighting scheme was therefore devised, assigning to the most prevalent hazard, aircraft vibration, a weight of ten, and to the least prevalent hazards, bird strikes for example, weights of one. The remaining environmental hazards were assigned integer values between one and ten based on their average relative frequency of occurrence.

In his assessment of damage potential using the original technique, the analyst was required to check off the environmental hazards to which the given structure would be exposed in service and to rate the level of exposure to each hazard as low, moderate or high, based on the location of the structure in the aircraft and the degree of protection that it receives relative to that hazard. Numerical weights were assigned to each of the three hazard exposure ratings. A damage potential score was then derived as a product of the hazard frequency and hazard exposure ratings.

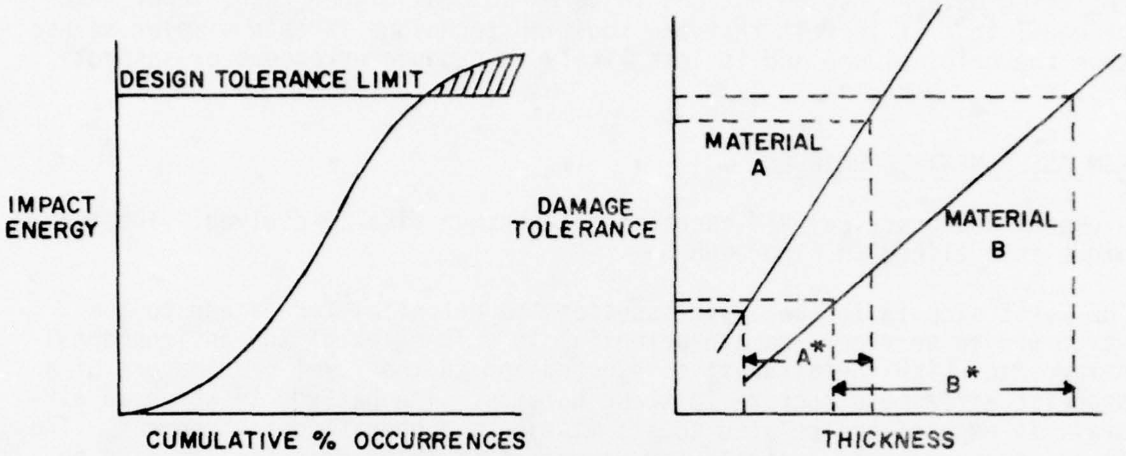
When the method was applied to various types of structural designs, the results often appeared inconsistent and unrealistic. Also, although the damage potential numbers were intended only to pinpoint possible areas of concern, they began to be interpreted as failure rates, and this made them appear even more unrealistic. Adjustments to the weighting values were tried, but this only produced distortions of other kinds.

The problem with the numerical scoring approach carried over to other areas of the R&M analysis. In assessing material factors related to reliability,

TABLE 45. SUMMARY OF ORIGINAL R&M ANALYSIS TECHNIQUE SCORING AND WEIGHTING SCHEMES

R&M Variable/Design Characteristic	Scoring or Weighting Scheme
Environmental Hazards	10 = most prevalent hazard 1 = least prevalent hazard 2-9 = intermediate values
Level of Exposure to Hazards	1 = low level of exposure 2 = moderate level of exposure 3 = high level of exposure
Damage Potential	Product of environmental hazard weight and level of exposure weight; summed to yield score by damage mode.
Damage Tolerance of Materials	10 = most damage tolerant material < 10 = lower damage tolerant materials Assigned to aluminum sheet, composite laminates and core materials for specific damage modes, based on characteristic mechanical properties
Reliability Rating - Material Factors	Product of damage potential score and damage tolerance weights; summed to yield score by damage mode.
Reliability Rating - Design Factors	10 = most positive attribute
Maintainability Rating - Design Factors	1 = least positive attribute
Maintainability Rating - Maintenance Factors	2-9 = intermediate values

for example, numerical values were developed to represent the relative damage tolerance of various materials based on specific mechanical properties. These values were then applied to the damage potential estimates to assess relative improvements or degradations in reliability. The intent was to assess the degree to which the choice of material had the potential for reducing or increasing the frequency of in-service damage or failure. But when the technique was applied to actual structures designs, the results produced often appeared to indicate variations in potential reliability that conflicted with engineering judgement or known experience. It was recognized that differences in ply orientation can drastically affect damage tolerance, and the possible combinations of these properties were much too numerous to evaluate. Figure 67 shows that two materials each having different damage tolerance characteristics can be equally acceptable in a given application depending on the thickness used.



* RANGE OF PRACTICAL DESIGN VALUES

Figure 67. Effect of Material Thickness on Damage Tolerance

The weighting and scoring methods applied to the other R&M attributes suffered similar problems and, when the products of the individual analyses were combined, a plausible conclusion about the R&M of a design concept could rarely be drawn. The following is illustrative of the problem:

Hazard Frequency	x	Hazard Exposure	x	Damage Tolerance	=	Rating
↑		↑		↑		↑
Range of Uncertainty (High)		Range of Uncertainty (High)		Range of Uncertainty (High)		Range of Uncertainty $(\text{High})^3$

The uncertainty associated with individual ratings, when combined, can produce results that are in error by two orders of magnitude.

It was concluded that the quantitative approach not only did not produce objective results but that the use of numerical measures implied a degree of precision not inherent to the analysis. The R&M analysis technique was accordingly modified. While addressing all of the same environmental factors and design variables as before, the revised technique described in the following pages requires only that the analyst make a series of simple qualitative observations and judgments in his assessment of a design, differentiating between design options in terms such as higher than, lower than or equal to. It is felt that the modified technique is both simpler to use than the original one and is less likely to produce erroneous or suspect results.

R&M ASSESSMENT TECHNIQUE

A useful and practical R&M assessment technique finally evolved. The technique is outlined in Figure 68.

The first step in the analysis assesses the potential for damage to the structure in service. Damage potential is a function of the environmental hazards to which the aircraft is exposed and to the level of exposure of a specific airframe structure to these hazards. The hazards to which an aircraft is exposed are related to its mission and operating environment. The level of exposure of a particular structure is related to its location on the aircraft and the degree of protection it receives from various hazards. These factors are systematically evaluated to arrive at an estimate of damage potential.

The next step in the analysis assesses the damage tolerance of the structure. A structure's tolerance to damage of various types is related to the properties of the materials used in its construction and to the presence or absence of specific design characteristics that tend either to worsen or lessen the degree of damage it sustains. The damage tolerance of the structure is rated relative to nine specific damage modes.

In the next step of the analysis, the likelihood of specific types of damage occurring in service is assessed. This is based on the potential for damage of each type and the damage tolerance of the structure as determined by the prior two steps in the analysis. The results are used to rate the overall structural reliability of the design. The hardware reliability of the design is rated separately, based on the number and types of fasteners used and such factors as vibration environment and load intensity.

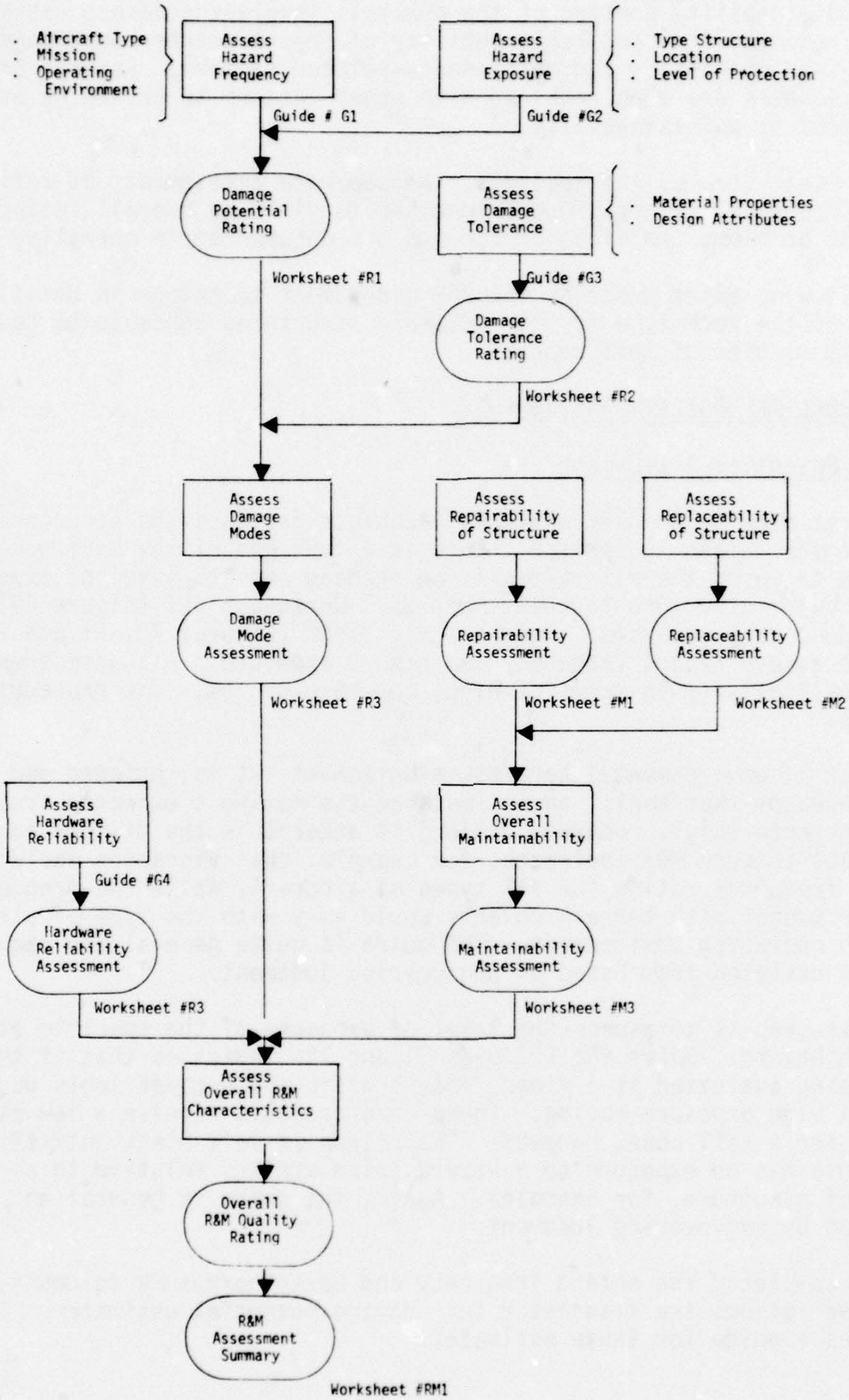


Figure 68. R&M Assessment Technique

The maintainability portion of the analysis involves separate assessments of the repairability and replaceability of the structure based on specific design characteristics and maintenance-related factors. Ratings of these two attributes are then combined with other factors to arrive at an overall assessment of maintainability.

In the final step of the analysis, the separate assessments of reliability and maintainability are brought together to yield an overall rating of R&M. Specific problems and areas of concern are documented in narrative form.

The following pages describe the R&M assessment technique in detail. Application of the technique to four advanced structures concepts is covered in the next section of this report.

STRUCTURAL RELIABILITY ASSESSMENT

Damage Potential Assessment

The first step in the R&M analysis technique assesses the structure's potential for damage in service. This is a function of the environmental hazards to which the aircraft will be exposed and the level of exposure of a particular structure to these hazards. Worksheet #R1 (Figure 69) is used to conduct this analysis. Guides #G1 and #G2 (Figures 70, 71 and 72) are used to assess hazard frequency and hazard exposure. All assessments are made qualitatively in terms of high, moderate or low. The procedure is as follows.

The list of environmental hazards in Worksheet #R1 is reviewed and for those hazards that apply, an estimate of the hazard's expected frequency of occurrence (high, moderate or low) is entered in the designated column. Guide #G1 (Figure 70) indicates, for example, that vibration would be given a high frequency rating for all types of aircraft, while the frequency rating for impact with terrain objects would vary with the type of aircraft and its operating environment. The guide is quite general and can be modified or deviated from based on engineering judgment.

The next step is to assess the level of exposure of the specific structure to each hazard. Guide #G2 (Figures 71 and 72) indicates that if the structure being evaluated is a floor, foot traffic and dropped tools would receive a high exposure rating. These hazards would receive a low exposure rating for a tail cone, however. The column is left blank entirely if the structure has no exposure to a hazard (bird strikes relative to an interior piece of structure, for example). Again, the guide is general and may be tempered by engineering judgment.

Having completed the hazard frequency and hazard exposure columuns, the respective ratings are translated into damage potential estimates. Figure 73 provides a guide for these estimates.

WORKSHEET #R1
DAMAGE POTENTIAL ASSESSMENT

Environmental Hazard	Hazard Frequency	Hazard Exposure	Damage Potential
Vibration			
Airborne Particles/F.O.D.			
Foot Traffic			
Dropped Tools/Parts			
Dropped/Shifting Cargo/Stores			
Door Slamming			
Rough Handling			
Bird Strikes			
Impact with Terrain Objects			
Work Stands/ Ground Vehicles			
Ballistic Impacts			
Corrosive Elements			

Rate for Type Aircraft,
Mission & Environment
(See Guide # G1)

Rate for Type Structure,
Location & Protection
(See Guide # G2)

Figure 69. Damage Potential Assessment Worksheet

GUIDE #G1
GUIDE TO ASSESSING HAZARD FREQUENCY

Environmental Hazard	Frequency Rating	Aircraft Type				Environment
		Utility	Attack	Observation	Cargo	
Vibration	High	X	X	X	X	All
Airborne Particles/ F.O.D.	Moder. Moder. High	X X 	X X	X X	X X 	Non-Combat Combat Combat
Foot Traffic	High	X	X	X	X	All
Dropped Tools/Parts	Moder.	X	X	X	X	All
Dropped/Shifting Cargo	Low Moder. High	 X 	X 	X 	 X	All
Door Slamming	Moder.	X	X	X	X	All
Rough Handling	Low	X	X	X	X	All
Bird Strikes	Low	X	X	X	X	All
Impact with Terrain Objects	Low Moder. High	X X 	X X	X X	X X 	Non-Combat Combat Combat
Work Stands/ Ground Vehicles	Low	X	X	X	X	All
Ballistic Impacts	Zero Moder. High	X X 	X X	X X	X X 	Non-Combat Combat Combat
Corrosives	Low Moder. High	X X X	X X X	X X X	X X X	Desert Average Salt Water

Figure 70. Guide to Assessing Hazard Frequency

GUIDE #G2

GUIDE TO ASSESSING HAZARD EXPOSURE (1 OF 2)

Hazard	Structure/Component	Location/Orientation	Level of Exposure
Vibration	Mechanically Fastened Fairings	Empennage	Heavy
		Main Rotor Pylon	Moderate
	Fuselage Joints & Splices	Tail Section	Moderate
		Mid-Fuselage Transmission Supports	Moderate
Airborne Particles/ F.O.D.	Cockpit Canopy	Frontal Area	Moderate
	Engine/Transmission Nacelles	Frontal Area	Moderate
	Vertical Pylon	Leading Edge	Moderate
	Horizontal Stabilizer	Leading Edge	Moderate
	Tail Cone	Horizontal Surfaces in High Velocity Downwash	Moderate
	Lower Fuselage	Horizontal Surfaces in High Velocity Updraft	Heavy
Foot Traffic	Floors		Heavy
	Engine Decks/Service Decks		Heavy
	Work Platforms		Heavy
	Roof Structure		Moderate
	Fairing/No-Step Structure	Horizontal Surfaces Near Walkways	Moderate
	Fuselage Skin Panels	Vertical Surfaces in Area of Fuselage Steps	Heavy
Dropped Tools and Parts	Floors		Moderate
	Engine Decks/Service Decks		Heavy
	Work Platforms		Heavy
	Roof Structure	Under Rotor Head	Moderate
	Fairing and Cowling	Under Rotor Head	Moderate
	Equipment Bay Shelves		Moderate
	Pylons, Pods	Lower fuselage under rotors and work platforms; horizontal surfaces	Moderate

Figure 71. Guide to Assessing Hazard Exposure (1 of 2)

Guide #G2
GUIDE TO ASSESSING HAZARD EXPOSURE (2 OF 2)

Hazard	Structure/Component	Location/Orientation	Level of Exposure
Dropped/ Shifting Cargo/Stores	Cargo Floors and Door Sills		Heavy
	Cargo Compartment Bulkheads		Heavy
	Cargo Doors (Interior)		Heavy
	Ammo Bay Floors and Walls		Heavy
Door Slamming	Crew Doors		Heavy
	Cargo Doors		Heavy
	Engine Access Doors		Moderate
	Equipment Bay Doors		Moderate
Rough Handling/ Dropped Structure	Removable Fairing, Cowling & Covers	Especially in awkward areas on upper fuselage	Heavy
Bird Strikes	Cockpit Canopy	Frontal area	Heavy
	Engine/Transmission Nacelles	Frontal area	Moderate
	Tail Pylon	Leading Edge	Moderate
	Horizontal Stabilator	Leading Edge	Moderate
Impact with Terrain Objects	Main Fuselage	Underside	Heavy
	Tail Cone	Underside	Heavy
	Horizontal Stabilizer		Moderate
	Pods and Pylons		Moderate
Impact with Work Stands & Ground Vehicles	Fuselage	Maximum projection of curved surfaces	Moderate
	Horizontal Stabilizer		Heavy
	Sponsons, Pods and Pylons	Protruding from aircraft	Heavy
Ballistic Impact	Forward Fuselage	Lower	Heavy
	Forward Fuselage	Upper	Moderate
	Center Fuselage	Lower	Heavy
	Center Fuselage	Upper	Moderate
	Rear Fuselage	Lower	Moderate
	Rear Fuselage	Upper	Light
Corrosive Elements	Fuselage Tub Areas	Areas of moisture entrapment	Heavy
	Interior Compartments		Moderate
	Interior of Pods & Pylons		Moderate

Figure 72. Guide to Assessing Hazard Exposure (2 of 2)

		Hazard Frequency		
		High	Moderate	Low
Hazard Exposure	High	High	High	Moder.
	Moder.	High	Moder.	Low
	Low	Moder.	Low	
		Damage Potential		

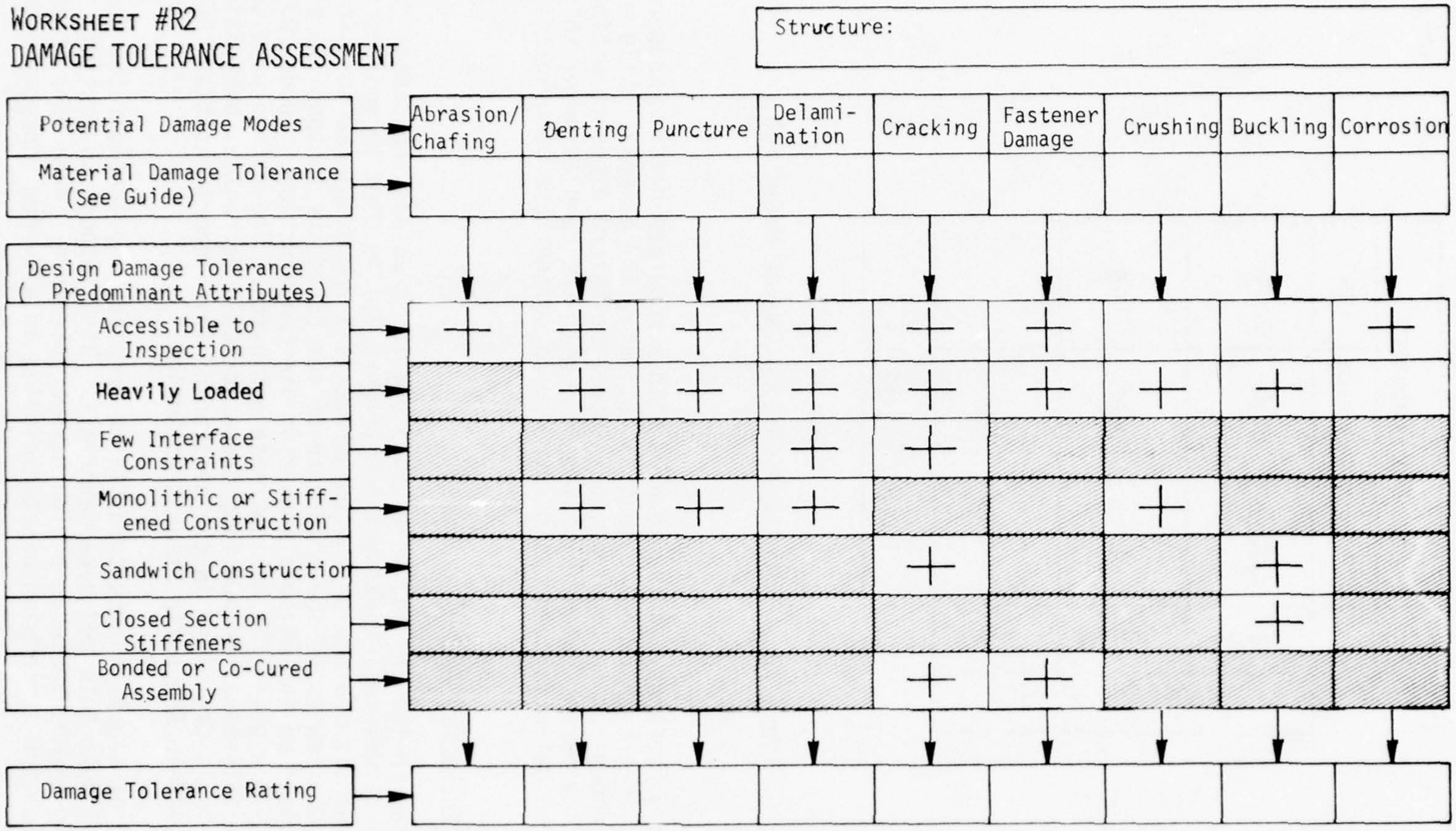
Figure 73. Guide to Assessing Damage Potential

A high hazard frequency rating coupled with a moderate hazard exposure rating results in a high damage potential estimate, for example, while a low hazard frequency rating coupled with a moderate hazard exposure rating results in a low damage potential estimate. With the completion of this assessment, the potential sources of structural damage have been identified and ranked.

Damage Tolerance Assessment

The next step in the R&M analysis procedure is to assess the damage tolerance of the structure. Worksheet #R2 (Figure 74) is used to conduct this analysis. Across the top of the worksheet is a row of blocks into which are entered estimates of the tolerance of the structure to specific modes of damage based on the materials used in its construction. Guide #G3 (Figure 75) is a guide for making these estimates. The method used to derive damage tolerance ratings from characteristic mechanical properties of the materials is described in the section of this report entitled "Reliability Factors in Composite Structures Design."

In the case of sandwich structure, it is necessary to consider the damage tolerance of both skin and core materials. With respect to denting, for example, the composites show a high damage tolerance. When used as the skin material for a sandwich panel employing an aluminum honeycomb core,



WORKSHEET #R2
DAMAGE TOLERANCE ASSESSMENT

Structure:

	Abrasion/ Chafing	Denting	Puncture	Delamination	Cracking	Fastener Damage	Crushing	Buckling	Corrosion
Potential Damage Modes									
Material Damage Tolerance (See Guide)									
Design Damage Tolerance (Predominant Attributes)									
Accessible to Inspection	+	+	+	+	+	+			+
Heavily Loaded	▨	+	+	+	+	+	+	+	
Few Interface Constraints	▨	▨	▨	+	+	▨	▨	▨	▨
Monolithic or Stiffened Construction	▨	+	+	+	▨	▨	+	▨	▨
Sandwich Construction	▨	▨	▨	▨	+	▨	▨	+	▨
Closed Section Stiffeners	▨	▨	▨	▨	▨	▨	▨	+	▨
Bonded or Co-Cured Assembly	▨	▨	▨	▨	+	+	▨	▨	▨
Damage Tolerance Rating									

Figure 74. Damage Tolerance Assessment Worksheet

Guide #G3

GUIDE TO ASSESSING DAMAGE TOLERANCE

	Composites and Aluminum					Core Materials		
						Honeycomb		Structr'l Foam
Type of Damage	Kevlar	Graphite	Fiber-glass	Boron	Aluminum	Alum.	Nomex	
Abrasion	Low	Moder.	Moder.	High	High	*	*	*
Denting	High	High	High	High	Low	Low	Moder.	High
Puncture	Low	Low	Low	Moder.	Moder.	*	*	*
Delamination	Low	Low	Low	Moder.	*	*	*	*
Cracking	Moder.	Low	High	High	High	High	Low	Low
Fastener Damage	Low	Moder.	Low	High	Moder.	*	*	*
Crushing	Low	Moder.	Low	High	High	Moder.	Moder.	Moder.
Buckling	Low	Moder.	Low	High	Moder.	*	*	*
Corrosion	*	*	*	*	Low	Low	*	*

*Mode not applicable.

Note: Tolerance rating may be affected by material thickness and ply orientation.

Figure 75. Guide to Assessing Damage Tolerance

the honeycomb's low tolerance to denting would prevail, however. Subjected to impact, the bond between the skin and core will normally prevent the skin from returning to its original shape as it does in monolithic form, leaving a dent in the panel. For sandwich construction it is also necessary to consider the possibility of internal damage not evident at the surface.

Design is the second factor that will affect the damage tolerance of a structure in service. In the section of this report entitled "Reliability Factors in Composite Structures Design" the significant design attributes affecting reliability and the nature of their effects were described.

The left-most column of Worksheet #R2 lists the design attributes having a potential influence on the damage tolerance of a structure. Design factors may be viewed as having either positive or negative effects in this respect. Monolithic construction, for example, has a positive influence with respect to the potential for denting, puncture and delamination. The approach taken is to identify the positive influences of various design attributes relative to damage tolerance and to assess the degree to which these attributes will enhance the damage tolerance inherent in the materials.

Worksheet #R2 presents a matrix of design attributes and damage modes. A plus sign at the intersection of a row and column indicates that the design attribute has a potentially mitigating influence on the damage mode. Shaded blocks indicate no influence or negligible influence with respect to that type of damage.

The procedure is to read down the list of reliability design factors and to check those that are predominant in the design. If the structure is made up entirely or primarily of flat panels, this factor would be checked. If the structure is comprised mostly of curved panels, "flat panels" would not be checked.

The final step in the analysis of damage tolerance is to weigh design attributes and material properties to arrive at an overall damage tolerance rating for each damage mode. Again, judgment is important. Assume for example that the structure receives a low tolerance rating for abrasion damage and that accessibility to inspection is checked as a predominant design attribute. The decision to be made is whether the ability to inspect for and detect abrasion in its early stages will effectively prevent abrasion from becoming a serious type of damage. If this is the judgment, the damage tolerance rating for abrasion would probably be elevated from a low rating based on material factors alone to a moderate or high rating based on the accessibility design factor. Similar reasoning is required for the assessment of damage tolerance relative to the other damage modes. When the row of blocks across the bottom of the worksheet has been completed, an assessment of the structure's tolerance to each type of damage will have been made.

Damage Mode Assessment

The next step in the R&M analysis is to estimate the relative probability of occurrence of the various damage modes. This is a function of damage potential and damage tolerance as developed in the first two steps of the analysis. Worksheet #R3 (Figure 76) is used to develop these estimates.

Damage tolerance ratings from Worksheet #R2 are transferred to the row of blocks across the top of the worksheet and damage potential ratings from Worksheet #R1 to the column of blocks to the left of the worksheet. Damage mode probability of occurrence is a coupling of these two factors; Figure 77 provides a guide.

As indicated by the guide, a high potential for damage coupled with a low tolerance for damage yields a high probability of damage. A low potential for damage coupled with a high tolerance for damage, on the other hand, indicates a small or negligible probability of damage.

The damage mode assessment documented in Worksheet #R3 is a key part of the analysis. It is essentially a checklist for reliability which, in addition to providing a comparative rating of designs, will highlight specific areas of concern. These are recorded in narrative form at the conclusion of the analysis, along with R&M concerns surfaced by other areas of the assessment, and serve as a basis for design improvement recommendations. When evaluating competing designs the best perspective will be maintained if specific design parameters are evaluated individually for all candidates. In this manner, judgments as to good or bad are tempered by the relative merits of the available design options.

HARDWARE RELIABILITY ASSESSMENT

The reliability assessment to this point has considered only the structure itself. The other aspect of reliability to be assessed is that of the associated hardware, i.e., mechanical fasteners and such items as hinges and latches. The reliability of the hardware is assessed separately, since it is largely independent of whether the structure is made of metals or composites, depending only on the methods of assembly and installation.

In an earlier section of this report covering service experience with helicopter airframes it was shown that failure of fasteners and other common hardware accounts for a large part of the unscheduled maintenance with present-day metal structures. This is an area where composites, owing to their monolithic form of construction, have the potential for significantly reducing maintenance.

The frequency of hardware failures is related primarily to the numbers and types of fasteners and other items of hardware used in the design. The vibration environment and the load intensity may also be factors. Heavy vibration will tend to increase the frequency of hardware-related failures,

WORKSHEET #R3
DAMAGE MODE ASSESSMENT

Structure:

Damage Tolerance Rating from Worksheet #R2 →

Damage Potential Rating from Worksheet #R1 →

Environmental Hazard	Damage Potential	Abrasion/ Chafing	Denting	Puncture	Delamination	Cracking	Fastener Damage	Crushing	Buckling	Corrosion
Vibration			▨	▨				▨	▨	▨
Airborne Particles/ F.O.D.							▨	▨	▨	▨
Foot Traffic				▨			▨			▨
Dropped Tools/ Parts		▨					▨			▨
Dropped/Shifting Cargo/Stores							▨			▨
Door Slamming		▨	▨	▨				▨		▨
Rough Handling		▨								▨
Bird Strikes		▨					▨			▨
Impact with Terrain Objects		▨					▨			▨
Work Stands/ Ground Vehicles		▨					▨			▨
Ballistic Impacts		▨	▨				▨	▨	▨	▨
Corrosives		▨	▨	▨	▨	▨	▨	▨	▨	

Damage Mode Assessment →

Figure 76. Damage Mode Assessment Worksheet

		Damage Potential		
		High	Moder.	Low
Damage Tolerance	Low	High	High	Moder.
	Moder.	High	Moder.	Low
	High	Moder.	Low	
		Damage Mode Assessment		

Figure 77. Guide to Assessing Damage Modes

and lightly loaded, lightly constructed components will be most affected by vibration. Guide #G4 (Figure 78) is an aid to rating hardware reliability. As indicated, a heavy structure having few permanent-type fasteners and located in a low vibration environment would receive a very good rating, whereas a light structure having many removable-type fasteners and located in a high vibration environment would receive a very poor rating.

MAINTAINABILITY ASSESSMENT

Repairability Assessment

The maintainability of an airframe structure, especially that of relatively permanent primary structure, is largely determined by its repairability. Simple economical repair reflects good maintainability while complex costly repair reflects poor maintainability. Repairability is affected by the types of repair the structure will require in service and the ease with which they can be made. Worksheet #M1 (Figure 79) is used to assess these factors.

Three types of repair are defined: a standard field repair, a complex repair and a custom-engineered repair. A checklist of factors is used to assess which of three types of repair a structure will likely require or, in some cases, to establish that no repair is possible. The procedure is to read down the list of factors and check those that apply to the structure being evaluated. For complex structures, it may be necessary to evaluate major sections of the structure independently.

GUIDE # G4
GUIDE TO ASSESSING HARDWARE RELIABILITY

Type Fasteners	Quantity	Vibration Environment	Load Intensity	Rating
None				Very Good
Permanent (Rivets, Lockbolts)	Few	Low	High	
			Low	
		High	High	Good
			Low	
	Many	Low	High	
			Low	Fair
		High	High	
			Low	
Removable (Screws, Bolts, Blind Fasteners)	Few	Low	High	
			Low	
		High	High	Poor
			Low	
	Many	Low	High	
			Low	Very Poor
		High	High	
			Low	

Figure 78. Guide to Assessing Hardware Reliability

WORKSHEET #M1
REPAIRABILITY ASSESSMENT

Structure:

		Types of Repair			
	Factor	Standard Field Repair	Complex Repair	Custom-Engineered Repair	No Repair
Design Related	Load Intensity	Light to Moderate ☐	Moderate to Heavy ☐	Heavy ☐	Heavy ☐
	Shape/Contour	Flat/Single Curvature ☐	Single/Double Curvature ☐		Complex Shape/ Contour/ Buildup ☐
	Interface Constraints	Few ☐	Some ☐	Many ☐	
	Skin/Web Form	Monolithic Sandwich ☐	Integrally Stiffened Sheet ☐		
	Stiffener/ Frame Form	Open Section ☐	Closed Section ☐		
Maintenance Related	Repair Materials	Stock/ Bulk Items ☐	Special Kits ☐	Special Storage/ Handling ☐	
	Environmental Requirements	Field Environment ☐	Controlled Environment ☐	Clean Room Conditions ☐	
	Tools and Equipment	Standard Field Type ☐	Special Field Type ☐	Factory Type ☐	
	Personnel Skills	Low Skill Level ☐	Intermediate Skill Level ☐	High Skill Level ☐	

Typical Component	Aircraft Skin/Fairing	Intermediate Frame	Longeron	Transmission Support Beam

This Structure Will Require Primarily →	Standard Field Repair ☐	Complex Repair ☐	Custom Engineered Repair ☐	Non-Repairable ☐

Figure 79. Repairability Assessment Worksheet

When the checklist has been completed it is reviewed and a determination is made of the type of repair that will be most prevalent. A majority of checks in boxes to the left of the matrix are compatible with standard field repair, while those to the center and right favor the other classes of repair or no repair. A lightly loaded flat structure with few interface constraints that can be repaired in a field environment by a person of low skill using commonly available tools and materials lends itself to standard field repair. Conversely, major damage to a complex, heavily loaded primary structure such as the transmission support beam would probably be unrepairable. Other types of structures suffering other degrees of damage will have a rated level of repairability between these extremes.

Replaceability Assessment

The second major factor contributing to the maintainability of airframe structures is the ease with which individual items of structure can be replaced, preferably in the field. Replaceability is assessed using Worksheet #M2 shown in Figure 80. Factors listed to the left of the matrix tend to indicate a simple field replacement while those to the right indicate a more difficult field replacement or a depot replacement of the structure.

Overall Maintainability Assessment

The overall maintainability of the structure is rated next using Worksheet #M3 shown in Figure 81. The rating is based on six factors including the repairability and replaceability factors assessed individually in the previous two steps. The maintainability of the structure is rated overall as good, fair or poor.

OVERALL R&M ASSESSMENT

At this point in the analysis the expected reliability and maintainability of the structure in service have been evaluated and rated. As the final step in the analysis an overall quality rating of structural reliability, hardware reliability and maintainability is made based on the results of the individual ratings. This represents the analyst's overall judgment of the design R&M. Figure 82 records this result. The next section of the report describes the method by which the overall R&M quality ratings are translated into estimates of life-cycle cost.

R&M Assessment Summary

The R&M analysis technique may be used in two ways: to compare the R&M attributes of alternative structures designs, especially those of composites versus metals, and to aid design by uncovering potential weaknesses and problem areas. For this second purpose, as the analyst proceeds through the analysis considering various aspects of reliability and maintainability, he may become aware of specific design problems relative to R&M; these should be documented. With respect to damage tolerance, for example,

WORKSHEET #M2
REPLACEABILITY ASSESSMENT

Structure:

Factor	Simple Field Replacement	Complex Field Replacement	Depot Replacement	No Replacement
Type of Joint	Simple Bolted Joint ☐	Semi-Permanent Fasteners ☐	Custom Fitted/ Shimmed ☐	Integral Molded/ Machined Structure ☐
Obstructions and Interfaces	Minor Parts and Components ☐	Major Components ☐	Major Components/ Plumbing/ Wiring ☐	
Jigs and Fixtures	None ☐	Field Type ☐	Factory Type ☐	
Spares	Small/ Inexpensive ☐	Large/ Inexpensive ☐	Large/ Expensive ☐	
Aircraft Downtime	Low ☐	Moderate ☐	Extensive ☐	

Typical Component	Fairing/ Door	Tail Cone	Rear Fuselage	Transmission Support Beam

Structure is →	Simple Field Replacement ☐	Complex Field Replacement ☐	Depot Replacement ☐	Non-Replaceable ☐

Figure 80. Replaceability Assessment Worksheet

WORKSHEET #M3
MAINTAINABILITY ASSESSMENT

Structure:

Factor	Good	Fair	Poor
Accessibility	Both Sides ☐	One Side ☐	Obstructed/ Inaccessible ☐
Inspectability	Visual ☐	Portable NDT ☐	Shop NDT ☐
Repairability (Worksheet #M1)	Standard Field Repair ☐	Complex Repair ☐	Custom-Engineered Repair ☐
Level of Repair	On Aircraft ☐	Field Shop ☐	Depot ☐
Replaceability (Worksheet #M2)	Easy Field Replacement ☐	Difficult Field Replacement ☐	Depot Replacement ☐
Expendability	Low Cost ☐	Moderate Cost ☐	High Cost ☐

Overall Maintainability Rating →	Good ☐	Fair ☐	Poor ☐

Figure 81. Maintainability Assessment Worksheet

Rating	Structural Reliability	Hardware Reliability	Maintainability
Very Good			
Good			
Fair			
Poor			
Very Poor			

Figure 82. Overall R&M Quality Rating

the poorest rating allowed by the assessment technique is "low", but the analyst may conclude from his evaluation of the design that the damage tolerance is so low as to be unacceptable. The specific concern should be documented for resolution with the designer. Worksheet #RM1 (Figure 83) shows the format used to record major areas of concern.

WORKSHEET # RM1
R&M ASSESSMENT SUMMARY

Structure:

Structural Reliability ☐ Hardware Reliability ☐ Maintainability ☐ Overall ☐

Type and Location of Damage	Source	Expected Frequency	R&M Concern

Figure 83. R&M Assessment Summary Worksheet

LIFE-CYCLE COST ASSESSMENT

Cost of Maintenance for Current-Inventory Airframe Structures

In the section of this report covering service experience with airframe structures, it was shown that the UH-1 and CH-47 helicopters share remarkably similar experience with respect to the frequency of airframe structures maintenance and its contribution to total aircraft maintenance. This similarity extends also to the cost of airframe maintenance. Table 46 was developed from Army data published in References 18, 19 and 20. Different maintenance cost figures were quoted for the UH-1 in References 18 and 20; the higher value quoted in Reference 18 was used. Costs were adjusted to 1978 price levels using a 36-percent DoD cost escalation rate.

TABLE 46. AIRFRAME MAINTENANCE COST FACTORS FOR CURRENT-INVENTORY HELICOPTERS

	UH-1	CH-47
Total Aircraft Maintenance Cost Per Flight-Hour (Dollars)	355	1,360
Airframe Maintenance Cost Per Flight-Hour (Dollars)	18	64
Airframe Percent of Total Maintenance Cost	5.0	4.7
Approximate Airframe Weight (Pounds)	1,200	4,500
Airframe Maintenance Cost (Dollars/Pound/Flight-Hour)	.015	.014

---

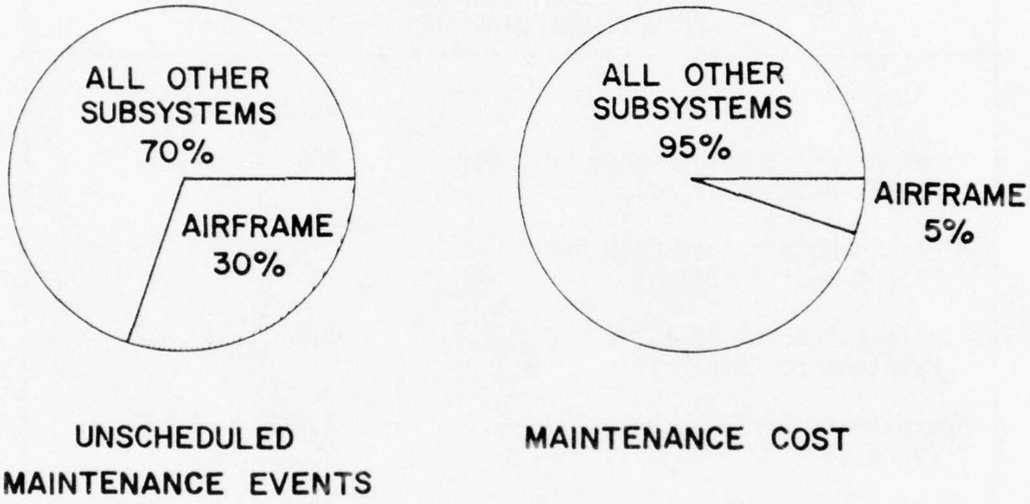
[18] Reddick, H. K., ARMY HELICOPTER COST DRIVERS, Report No. USAAMRDL-TM-7, U. S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Va., August 1975, AD A015517.

[19] EXECUTIVE SUMMARY REPORT, CH-47A ASSESSMENT AND COMPARATIVE FLEET EVALUATIONS, FINAL REPORT, USAAVSCOM Technical Report No. 74-46, U. S. Army Aviation Systems Command, St. Louis, Mo., November 1974.

[20] EXECUTIVE SUMMARY REPORT, UH-1H ASSESSMENT AND COMPARATIVE FLEET EVALUATIONS, USAAVSCOM Technical Report No. 75-3, U. S. Army Aviation Systems Command, St. Louis, Mo., April 1975.

These statistics indicate that 1.5 cents per pound per flight-hour can be used as a rough rule of thumb for the overall cost of airframe maintenance for current-inventory Army helicopters. This cost will vary widely for individual items of structure, of course, with light fragile structures being more costly per pound than average to maintain and heavy rugged structures less costly than average.

While the airframe system generates upward of one-third of the unscheduled maintenance events on the helicopter, the average cost of these maintenance events is quite low compared with other systems of the aircraft, as Figure 84 illustrates.



UNSCHEDULED MAINTENANCE EVENTS

MAINTENANCE COST

Figure 84. Frequency Versus Cost of Airframe Maintenance

Table 47 gives a representative breakdown of airframe maintenance costs for a present-day utility class helicopter. The table was derived from historical data and engineering judgment as follows: The unscheduled maintenance events per flight-hour and the distribution of these events among primary structure, secondary structure and hardware are approximate values for the UH-1 obtained from Reference 4. The total airframe maintenance cost of $18 per flight-hour, also an approximate value for the UH-1, was obtained from Reference 18 (adjusted to 1978 price levels) as was the 40 percent/60 percent apportionment of that cost to primary structure and secondary structure respectively.

TABLE 47. REPRESENTATIVE MAINTENANCE COST BREAKDOWN FOR A UTILITY CLASS HELICOPTER AIRFRAME

Structural Element	Maint. Events/ Flt-Hr.	Average Parts Cost	Average Labor Cost	Average Cost/ Event	Cost/ Flt-Hr.
Primary Structure					
Structure	.030	$69	$152	$221	$6.65
Hardware	.020	14	14	28	.55
Total/Average	.050	47	97	144	7.20
Secondary Structure					
Structure	.110	50	26	76	8.30
Hardware	.090	14	14	28	2.50
Total/Average	.200	33	21	54	10.80
Total Airframe	.250	$36	$36	$72	$18.00

The balance of the table was constructed as follows: Army published statistics contained in References 19, 20 and 21 indicate that the division of aircraft maintenance cost between parts and labor is approximately equal:

	Percent of Maintenance Cost		
	OH-58A	UH-1H	CH-47A
Total Airframe Parts	54.8	47.0	58.0
Total Airframe Labor	45.2	53.0	42.0

The total $18 per flight-hour airframe maintenance cost was thus equally divided among parts and labor. The same three Army reports for these three aircraft give a breakdown of the major contributors to airframe maintenance

---

[21] EXECUTIVE SUMMARY REPORT, OH-58A FLEET ASSESSMENT, USAAVSCOM Technical Report No. 75-34, U. S. Army Aviation Systems Command, St. Louis, Mo., September 1975.

cost, the majority of which in each case are items of secondary structure. This data indicates that parts replacement represents approximately two-thirds of secondary structures maintenance cost.

	Percent of Maintenance Cost		
	OH-58A	UH-1H	CH-47A
Secondary Structures Parts	68.7	60.7	76.6
Secondary Structures Labor	31.3	39.3	23.4

With all of the above data, it was necessary only to derive relationships for the cost of maintenance related to primary structure, secondary structure and airframe hardware that would preserve the approximate cost ratios developed from the historical data. The derived values for the cost of maintaining primary structure indicate that labor rather than replacement parts is the predominant cost element. This seems reasonable in view of the fact that little primary structure is replaceable in the field. The cost of parts and labor was assumed to be equal for maintenance related to airframe hardware. While apportionments other than the one shown in Table 47 could be derived, individual values could not differ significantly and still fit the historical experience. In Figure 85 the data from Table 47 is shown in terms of percentage contributions to total airframe maintenance cost.

Cost Analysis Method

Advanced composite structures for helicopters will be replacing structures of conventional metal design. The cost of maintaining the composite structure versus that of maintaining the metal structure will be largely a function of their relative R&M characteristics. The R&M assessment technique described in the preceding section allows the R&M characteristics of both types of structures to be evaluated and compared. The assessment is made in terms of the three attributes:

Structural Reliability

Hardware Reliability

Maintainability

Each of these attributes is rated qualitatively using one of five ratings:

Very Good

Good

Fair

Poor

Very Poor

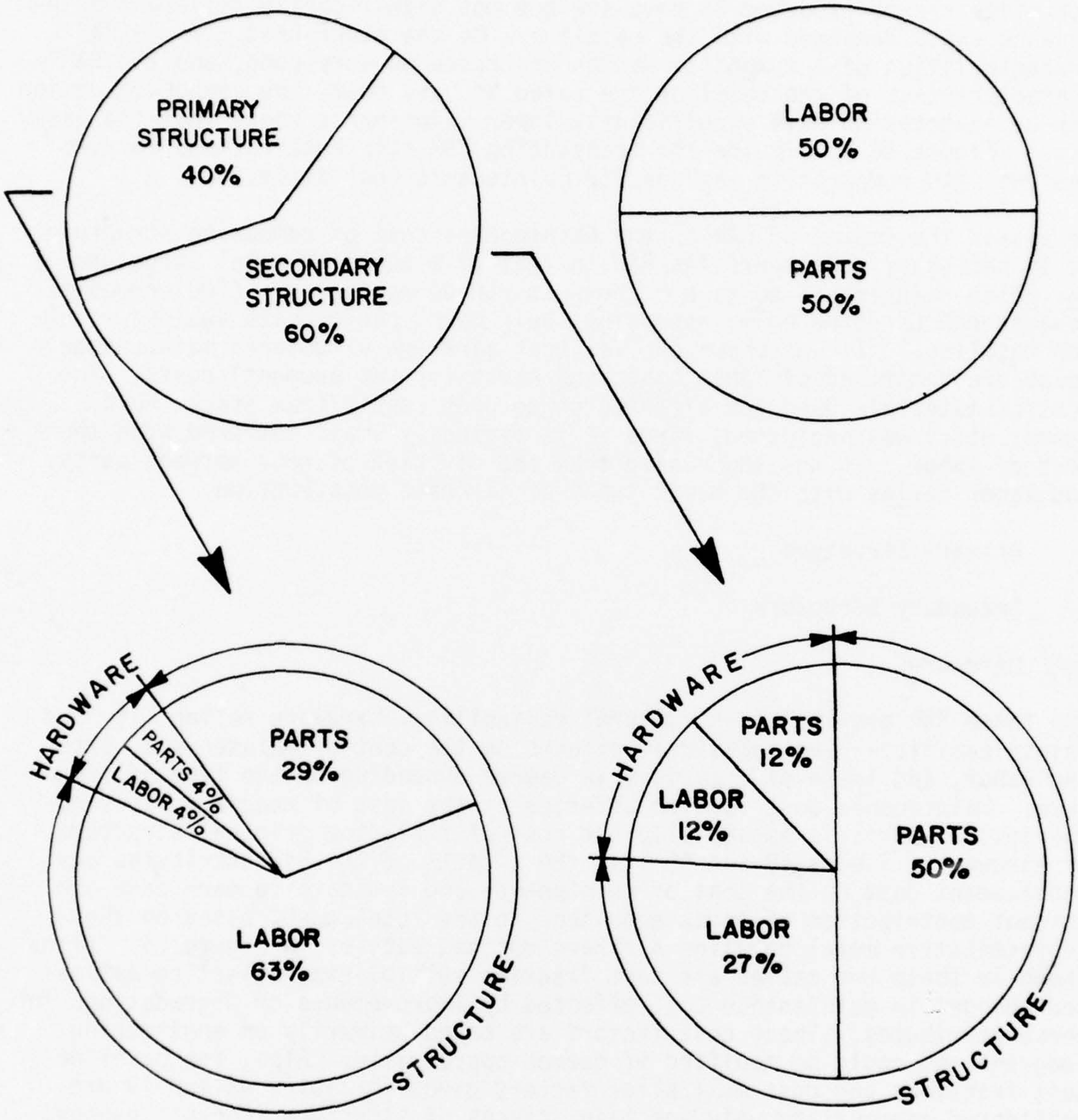


Figure 85. Division of Airframe Maintenance Costs

The potential that a composite structure has for reducing or increasing maintenance cost is a function of its R&M relative to the structure for which it is a replacement or an alternate. If the R&M characteristics of a composite design are rated as very good but the R&M characteristics of an alternate metal design (the baseline) are also rated as very good, the composite can be expected to have low but not significantly different maintenance costs compared with the baseline. On the other hand, if the R&M characteristics of a composite design are rated as very good, and the R&M characteristics of the baseline are rated as very poor, the composite design can be expected to have significantly lower maintenance costs than the baseline. Figure 86 is a guide for translating R&M attribute ratings for two designs into comparative ratings for maintenance cost analysis.

To assess the impact of R&M on the maintenance cost of composite structure, it is necessary to compare its R&M to that of a baseline metal structure for which maintenance costs are known or can be estimated. (Two composite designs can be compared by assessing their maintenance costs against a common baseline.) It was shown earlier that airframe structures maintenance costs are comprised of labor costs and parts (or replacement) costs. The cost of materials used for airframe structures repair (raw stock, bulk items, etc.) was neglected, since it is typically small compared with the cost of labor. It was shown also that the division of cost between parts and labor varies with the basic types of airframe construction:

Primary Structure

Secondary Structure

Hardware

The three R&M attributes--structural reliability, hardware reliability and maintainability--have individual effects on the cost of maintenance parts and labor, and these effects vary in degree depending on the type of structure. Maintenance cost is also affected by the cost of manufacture, since the cost of parts is essentially the cost of replacing original structure or hardware. Tables 48 and 49 list the effects of the R&M attributes and replacement cost on the cost of maintenance and indicate in each case the percent contribution of parts and labor to the total cost, based on the representative metal baseline airframe defined earlier in Figure 85. Also shown in these two tables are cost fraction multipliers reflecting estimated changes in maintenance cost effected by improvements or degradations in these attributes. These cost factors are based primarily on engineering judgment and could be modified if deemed appropriate. Also, the baseline cost fractions and cost multiplier factors given in Tables 48 and 49 are considered appropriate only for major pieces of structure (cockpit canopy, tailcone, etc.). For smaller components a much greater range of variability could be anticipated.

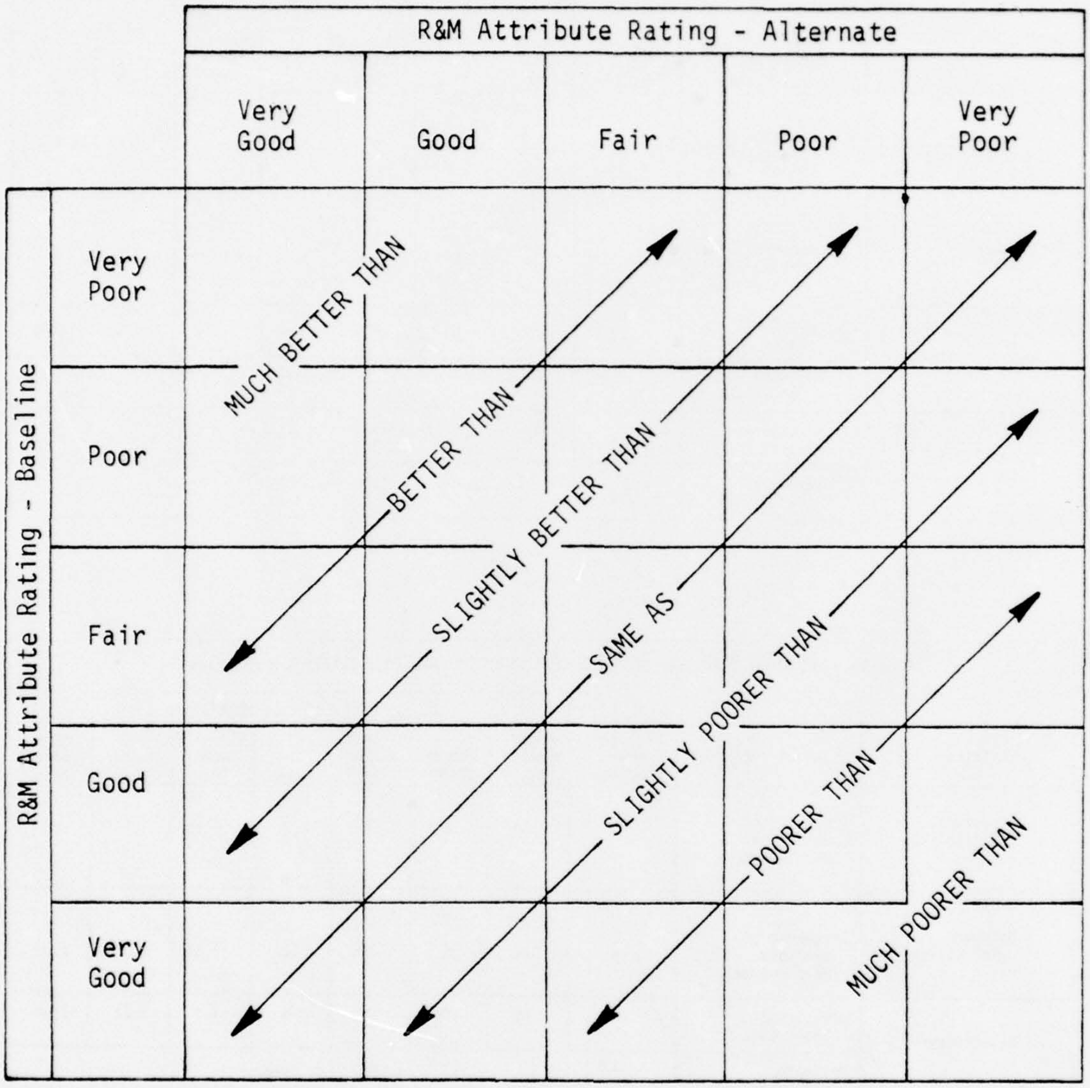


Figure 86. R&M Attribute Rating Matrix

TABLE 48. PRIMARY STRUCTURE BASELINE COST FRACTIONS AND COST FRACTION MULTIPLIERS

			Cost Fraction Multiplier						
Attribute	Effect of Attribute	Baseline Cost Fraction	Much Better Than	Better Than	Slightly Better Than	Same	Slightly Poorer Than	Poorer Than	Much Poorer Than
Structural Reliability	Frequency of Structural Maintenance	-	.50	.65	.80	1.0	1.25	1.5	2.0
Hardware Reliability	Frequency of Hardware Maintenance	-	.25	.5	.75	1.0	1.5	2.0	4.0
Maintainability	Structures Labor Cost	.63	.50	.65	.80	1.0	1.25	1.5	2.0
	Hardware Labor Cost	.04	1.0 (No Effect)						
Acquisition Cost	Structures Parts Cost	.29	Composite Cost ÷ Baseline Cost						
	Hardware Parts Cost	.04	1.0 (No Effect)						

TABLE 49. SECONDARY STRUCTURE BASELINE COST FRACTIONS AND COST FRACTION MULTIPLIERS

			Cost Fraction Multiplier						
Attribute	Effect of Attribute	Baseline Cost Fraction	Much Better Than	Better Than	Slightly Better Than	Same	Slightly Poorer Than	Poorer Than	Much Poorer Than
Structural Reliability	Frequency of Structural Maintenance	-	.50	.65	.80	1.0	1.25	1.5	2.0
Hardware Reliability	Frequency of Hardware Maintenance	-	.25	.5	.75	1.0	1.5	2.0	4.0
Maintainability	Structures Labor Cost	.27	.65	.80	.90	1.0	1.1	1.25	1.50
	Hardware Labor Cost	.12	1.0 (No Effect)						
Acquisition Cost	Structures Parts Cost	.50	Composite Cost ÷ Baseline Cost						
	Hardware Parts Cost	.12	1.0 (No Effect)						

Structural Reliability Cost Effect

For the reliability of primary structure, a range of .5 to 2.0 was chosen to represent the cost delta between a composite design that has much better R&M than a metal structure and one that has much poorer R&M than a metal structure. The reasoning here is that a good composite structure might reduce by one half the number of damage events that would be sustained by a metal design, but a reliability improvement greater than this would probably be beyond the state of the art or would involve unacceptable weight and cost penalties. Following the same reasoning, a poor composite design might double the number of damage events sustained by a metal structure, but a reliability degradation greater than this would be evident during qualification and would prevent the structure from being introduced to service or would require design improvement by the manufacturer.

Hardware Reliability Cost Effect

A much larger delta change effect (.25 to 4.0) was estimated for hardware reliability because bonded composite structures have the potential for drastically reducing (in some cases totally eliminating) hardware-related maintenance.

Maintainability Cost Effect

The effect of maintainability attributes on labor costs associated with airframe hardware maintenance is considered negligible because the introduction of composites will have no predictable effect on the installation or replacement of common hardware (fasteners, hinges, latches, etc.). Maintainability attributes may have a pronounced effect on the labor costs associated with the maintenance of the structures themselves, with primary structure estimated to be more affected by maintainability characteristics than secondary structure.

Many items of secondary structure (fairings, cowlings, etc.) on present-day helicopters are already constructed from composites. Repair techniques for these structures, both metals and composites, are well established and consist for the most part of simple patches. Additional items of secondary structure will be candidates for composites in the future, but there is no reason to believe that the repair of these structures will differ significantly from that of structures currently in service. New techniques and materials might be developed to improve the repair of all structures of this type, but there is no reason to expect that one structure can be made much more or much less repairable than another. This is probably true of replacement maintenance as well. It is doubtful that individual items of secondary structure, which are typically easy to replace anyway, can be made much more or much less replaceable than other items of the same type. Therefore, the influence of maintainability attributes on the cost of maintaining secondary structure is considered minimal, and a small delta change effect (.65 to 1.5) was assigned.

Maintainability has potentially a very large effect on the cost of maintaining primary structure, however. Few composite structures of this type have been placed in service, and repair techniques for them are just now beginning to be developed. Depending on the repair methods that are developed, maintenance costs for primary structure could vary substantially from maintenance costs for equivalent metal structure. The labor cost for replacement of primary structure could also be greatly affected by maintainability attributes, composites possibly being much easier or much more difficult to replace than metals, depending on the design. A large delta change effect (.5 to 2.0) was therefore selected to represent the influence of maintainability on maintenance labor costs for primary structure.

Life-Cycle Cost Estimating Procedure

A maintenance cost prediction for a composite structure is obtained from the R&M assessment of the design, the projected maintenance cost effects of the R&M attributes, and known or estimated maintenance costs for a baseline metal structure of the same type. The procedure is as follows.

The R&M assessment technique is used to assess the R&M characteristics of the metal baseline and the proposed composite design. Qualitative ratings of the three R&M attributes are obtained and converted to an R&M quality comparison using the R&M attributes rating matrix (Figure 86) as illustrated below:

R&M Attribute	Baseline Rating	Composite Rating	Quality Comparison
Hardware Reliability	Poor	Very Good	Much Better Than
Structural Reliability	Good	Fair	Slightly Poorer Than
Maintainability	Fair	Very Good	Better Than

Using Table 48 for primary structure and Table 49 for secondary structure, the baseline percentage breakdown for parts and labor costs is obtained, along with the cost fraction multipliers corresponding to the R&M quality comparison rating. The difference in acquisition cost of the proposed composite design versus the metal baseline is also calculated. An illustration is shown below:

R&M Attribute	Quality Comparison	Attribute Effect	Baseline Fraction	Cost Multiplier
Structural Reliability	Slightly Poorer Than	Frequency of Structural Maint.		1.25
Hardware Reliability	Much Better Than	Frequency of Hardware Maint.		.25
Maintainability	Better Than	Structures Labor Cost	.63	.65
		Hardware Labor Cost	.04	1.0
Acquisition Cost		Structures Parts Cost	.29	.7
		Hardware Parts Cost	.04	1.0

These factors are used to calculate a predicted delta change in maintenance cost for the composite versus the baseline:

$$\begin{aligned} MC_{\Delta} &= SR_{\Delta} \times (SPC_{\Delta} \times SPC + SLC_{\Delta} \times SLC) \\ &\quad + HR_{\Delta} \times (HPC + HLC) \end{aligned}$$

Where:

MC_{Δ} = Predicted Change in Maintenance Cost*

SR_{Δ} = Predicted Change in Structural Reliability*

SPC_{Δ} = Predicted Change in Structures Parts Cost*

SPC = Structures Parts Cost Fraction of Total Baseline Maintenance Cost

SLC_{Δ} = Predicted Change in Structures Labor Cost (Maintainability Effect)

SLC = Structures Labor Cost Fraction of Total Baseline Maintenance Cost

HR_{Δ} = Predicted Change in Hardware Reliability

*Expressed as a cost multiplier.

HPC = Hardware Parts Cost Fraction of Total Baseline Maintenance Cost

HLC = Hardware Labor Cost Fraction of Total Baseline Maintenance Cost

Using the illustration carried through the previous discussion, including the assumed change in acquisition cost of .7 (composite 30 percent less expensive than baseline), the predicted change in maintenance cost would be calculated as follows:

$$
\begin{aligned}
MC_{\Delta} &= SR_{\Delta} \text{ X } (SPC_{\Delta} \text{ X } SPC + SLC_{\Delta} \text{ X } SLC) \\
&+ HR_{\Delta} \text{ X } (HPC \; + HLC) \\
&= 1.25 \text{ X } \left[(.7)\ (.29) + (.65)\ (.63)\right] + \; .25\ (.04 + .04) \\
&= \; .79
\end{aligned}
$$

Based on this example, the composite structure is projected to have 21 percent lower maintenance costs than the equivalent metal baseline.

The predicted change in total life-cycle cost is a function of the predicted change in maintenance cost and the estimated change in acquisition cost:

$$LCC_{\Delta\$} = (MC_{\Delta} - 1) \text{ X } MC_{BASE} \text{ X Service Life} + (ACQ_{\Delta} - 1) \text{ X } ACQ_{BASE}$$

Where:

$LCC_{\Delta\$}$ = Predicted Change in Life Cycle Cost ($)

MC_{BASE} = Maintenance Cost of Baseline ($/Flight-Hour)

Service Life = Expected Service Life of Aircraft or Structure (Flight-Hours)

ACQ_{Δ} = Estimated Change in Acquisition Cost (Candidate/Baseline)

ACQ_{BASE} = Acquisition Cost of Baseline ($)

Assuming a baseline acquisition cost of $30,000, a baseline maintenance cost of $1.00 per flight-hour and an expected aircraft service life of 8,000 flight-hours, the example being followed would yield the following life-cycle cost delta:

$$
\begin{aligned}
LCC_{\Delta\$} &= (.79 - 1) \times 1.00 \times 8,000 \; + \; (.7 - 1) \times 30,000 \\
&= -\ \$1,680 - \$9,000 \\
&= -\ \$10,680 \text{ per aircraft}
\end{aligned}
$$

In this example the composite structures design saves an estimated $10,680 over the life of the aircraft compared with the equivalent metal baseline design. Approximately 15 percent of the saving occurs in maintenance cost and 85 percent in acquisition cost.

Application of the Method

Use of the R&M assessment technique and life-cycle cost estimating method described herein requires a definition of the advanced composites design (either conceptual or actual), including an estimate of its cost of acquisition, and the definition of an equivalent structure of conventional metal design (the baseline), including its acquisition cost and cost of maintenance. Alternate composites designs are compared by assessing each with respect to a common metal baseline.

It is assumed that the acquisition costs of the composites design and metal baseline will be known or can be estimated. The cost of maintenance for the metal baseline may not be known, however. In the absence of such data, Table 50 may be used to obtain representative per flight-hour maintenance costs for generic items of metal airframe structure for a utility class helicopter.

TABLE 50. REPRESENTATIVE PER FLIGHT-HOUR MAINTENANCE COSTS FOR PRESENT DAY UTILITY CLASS HELICOPTER AIRFRAME STRUCTURES*

		Dollars/Flight Hour
Primary Structure		
Cockpit Canopy		.42
Cockpit Structure		.51
Upper Fuselage		.55
Lower Fuselage		.82
Rear Fuselage		.68
Tail Cone		1.08
Tail Pylon		2.09
Stabilizer		1.05
	Total	7.20
Secondary Structure		
Floors		1.50
Fairing and Cowling		3.70
Aircraft Doors		3.64
Transparencies		1.92
	Total	10.80

* 10,000-pound weight class helicopter.

The table was derived by apportioning the published $18 per flight-hour cost of maintaining the UH-1 helicopter airframe on the basis of predicted failure rates for airframe structures of the UH-60A BLACK HAWK helicopter. It should be noted that the per flight-hour costs are estimates for all items of structure of each generic type and would have to be apportioned further to obtain an estimate of the cost for a single piece of structure of each type. Also, the costs are representative of utility class helicopters in the 10,000-pound weight class category and would have to be adjusted upward or downward for significantly larger or significantly smaller aircraft. The 1.5 cents per pound per flight-hour estimates for current-day airframes discussed earlier may be used for this purpose.

COST SENSITIVITY

The sensitivity of maintenance costs and life-cycle costs to changes in each of four major cost variables (structural reliability, hardware reliability, maintainability and replacement cost) was investigated.

Maintenance Cost Sensitivity

Using the historical cost breakdowns for maintenance of primary and secondary airframe structures given in Tables 48 and 49 and the maintenance cost formula given previously, the effect of each variable on maintenance cost was expressed as a ratio of the least influential variable, hardware reliability, as shown in Table 51. For example, the maintenance cost of primary structure is 3.6 times more sensitive to component replacement cost than it is to hardware reliability.

TABLE 51. MAINTENANCE COST SENSITIVITY

Cost Variable	Relative Effect on Maintenance Cost*	
	Primary Structure	Secondary Structure
Structural Reliability	11.5	3.2
Replacement Cost	3.6	2.1
Labor Cost (Maintainability Effect)	7.9	1.1
Hardware Reliability	1.0	1.0

*Normalized to hardware reliability effect.

Since it dictates the frequency at which the more significant types of damage and repair occur, structural reliability affects both the parts and labor components of cost and has the greatest influence on the cost of maintenance for both primary and secondary structure. The effect is less for secondary structure than it is for primary structure, because a larger part of secondary structures maintenance is related to hardware reliability.

For secondary structure replacement cost is the next most influential cost variable, while for primary structure labor cost (a function of maintainability) is next most influential. Secondary structures maintenance frequently involves the replacement of damaged structure (fairings, panels, etc.), whereas primary structures maintenance primarily involves the more labor-intensive repair of the structure in place. The cost of maintaining secondary structure can thus be reduced most readily by reducing the manufactured cost of the structure, while the cost of primary structures maintenance can be reduced more directly by improving the repairability (ease of repair) of the structure.

Hardware reliability has a minor influence on the maintenance costs of both primary and secondary structure, although it is more of a factor with secondary structure which typically experiences a larger number of hardware failures. While the potential impact on cost is small, composite structures have the potential for eliminating many nuisance-type maintenance actions by reducing the quantity of hardware.

Life-Cycle Cost Sensitivity

The life-cycle cost of an airframe structure is made up of the initial acquisition cost and the cost of maintenance. The manufactured cost of the structure represents 100 percent of the initial acquisition cost and some fraction of the cost of maintenance, depending on the number of times the structure is replaced because of damage or failure over the lifetime of the aircraft.

The manufactured cost of the structure as it affects the cost of replacement has a significant effect on the maintenance cost of airframe structures, especially secondary structures. When initial acquisition is also considered, manufacturing cost becomes the dominant factor in the life-cycle cost of most airframe structures as will be shown by the life-cycle assessment of four advanced structures in the next section of this report.

ASSESSMENT OF ADVANCED DESIGN CONCEPTS

The R&M/cost assessment technique described in the preceding section of the report was used to evaluate advanced composite designs for a selected group of helicopter airframe structures. Actually, development of the assessment technique involved an iterative process wherein the technique was applied to the evaluation of one or more of these designs, modified to correct deficiencies, and tested again. The four structures selected for detailed analysis represent the complete range of structures found in helicopter airframes and include the cockpit canopy, rear fuselage, stabilator and transmission support structure (Figure 87). In each case two advanced composite designs and a metal baseline design were evaluated and compared.

DESIGN CONCEPTS

Cockpit Canopy

The cockpit canopy design concepts are described in Table 52 and illustrated in Figures 88 and 89. Cockpit canopies constructed of fiberglass have been in service on the CH-53 helicopter since 1961. The UH-60A Black Hawk helicopter also has a fiberglass canopy, one that makes extensive use of unidirectional tapes for reinforcing purposes. Selection of the cockpit canopy provided the opportunity to evaluate the reliability and maintainability of a lightly loaded complex structure that has a substantial amount of actual service experience.

Stabilator

The stabilator design concepts are described in Table 53 and illustrated in Figures 90 and 91. The Sikorsky S-76 helicopter has an all-composite stabilator, and composite derivatives of the design have been investigated for the UH-60.

The stabilator provided the opportunity to evaluate a moderately expensive replaceable primary structure. Repairability is of interest since the stabilator contains members that are highly loaded, such as the spar and root fittings, as well as lightly loaded leading and trailing edge panels. The effect of repair on stiffness and dynamic characteristics is also of concern.

Rear Fuselage

The rear fuselage design concepts are described in Table 54 and illustrated in Figures 92 and 93. A lightly loaded primary structure, the rear fuselage is a prime candidate for redesign with composites because of the large potential for weight and cost savings. Lower manufacturing costs will be achieved primarily through a reduction in parts count and assembly man-hours. The conversion to larger monolithic pieces of structure that composites permit places great importance on the ability to repair structural components such as frames and beams in place in the field. Repair concepts developed for a large primary structure such as the rear fuselage would be generically applicable to a wide range of advanced composite structures. The modular

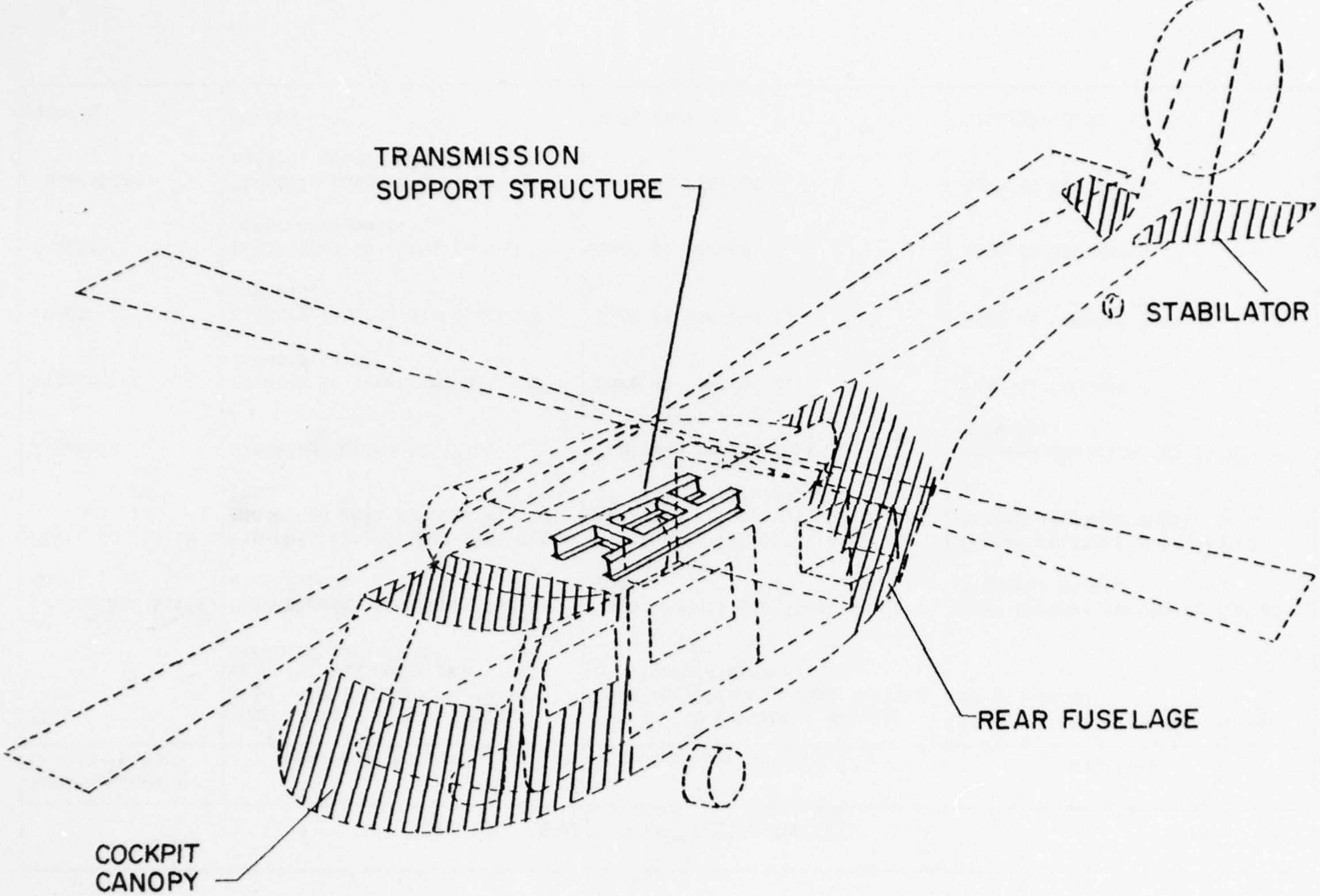


Figure 87. Advanced Structures Design Concepts

TABLE 52. COCKPIT CANOPY DESIGN CONCEPTS

Sub-Component/ Design Factor	Composite Concept I	Composite Concept II	Baseline
Skin	Monolithic fiberglass/epoxy skin. Small Nomex honeycomb pads for equipment mounting. Compound curvature.	Similar to Concept I except that Kevlar/epoxy used instead of fiberglass/epoxy.	Aluminum sheet stretch-formed or planished.
Frames and Stiffeners	Open channel sections; fiberglass/epoxy.	Open channel sections; Kevlar/epoxy.	Open channel sections; formed aluminum sheet.
Posts and Sills	Fiberglass/epoxy hat sections bonded to skin to form closed box.	Similar to Concept I except that Kevlar/epoxy used instead of fiberglass/epoxy.	Closed sections extruded or formed aluminum sheet.
Assembly	Skeleton bonded to skin.	Skeleton bonded to skin.	Riveted skeleton and skin assembly.
Attachment	Riveted to lower cockpit and forward cabin.	Same as Concept I.	Same as Concept I.
Loads	Lightly loaded; air loads are largest.	Same as Concept I.	Same as Concept I.
Access	Restricted by consoles and other equipment.	Same as Concept I.	Same as Concept I.
Interfaces	Throttle quadrant; electrical panels, windshield wipers.	Same as Concept I.	Same as Concept I.
Source	UH-60A	Reference 2	Sikorsky S-61

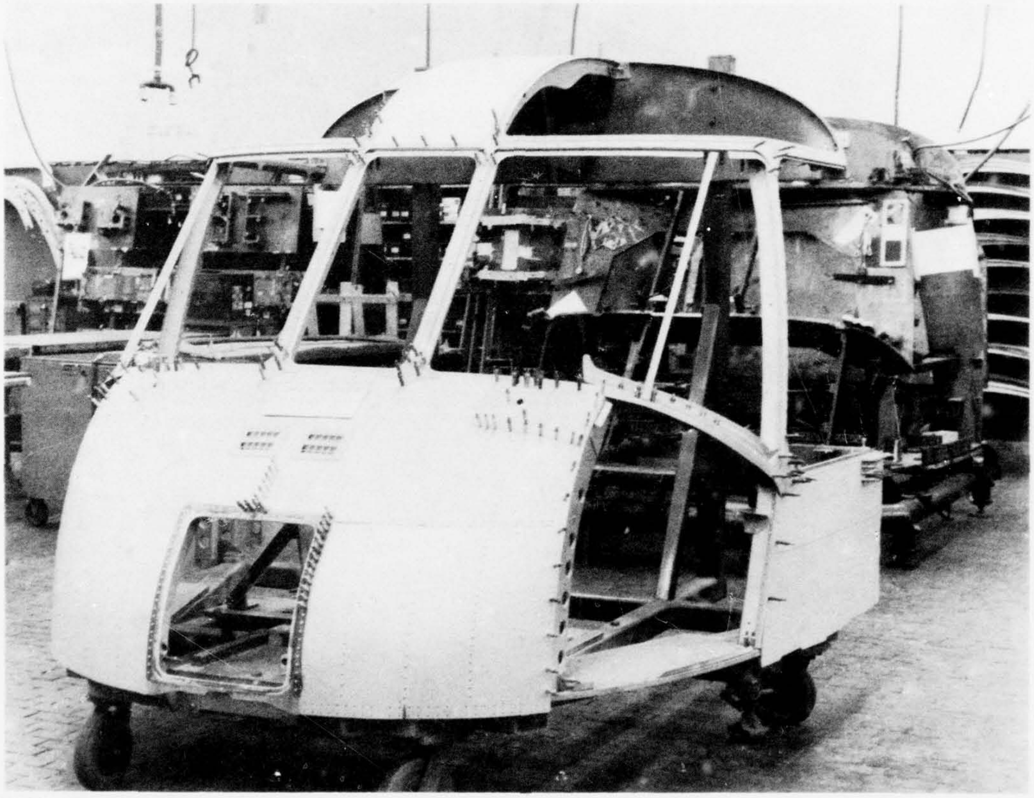


Figure 88. S-61 Cockpit Canopy

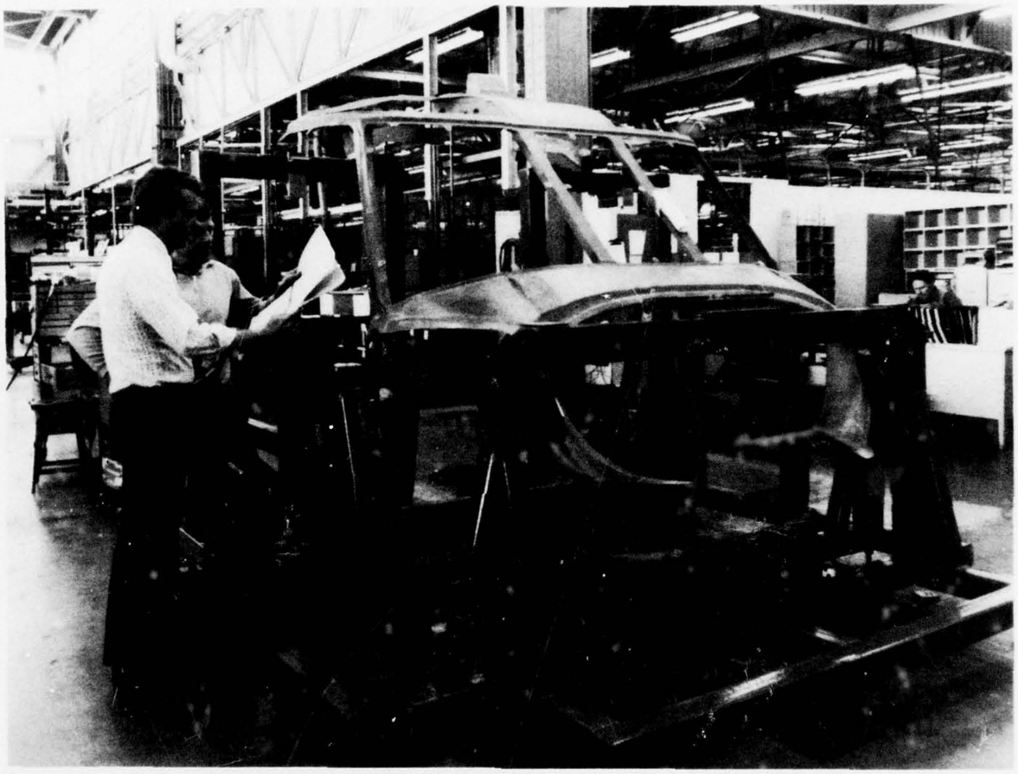


Figure 89. UH-60A Cockpit Canopy

TABLE 53. STABILATOR DESIGN CONCEPTS

Design Factor/ Sub-Component	Composite Concept I	Composite Concept II	Baseline
Skin	Monolithic Kevlar/epoxy supported by Nomex honeycomb forward and aft of the spar.	Thin graphite/epoxy monolithic laminates.	Aluminum sheet.
Spar	Kevlar/epoxy closed box beam with graphite/epoxy plies interlayered for cap material.	Graphite/epoxy, bead stiffened I-beam, monolithic, with flanges for skin rivets.	Built-up riveted aluminum sheet and extrusions.
Ribs	None. (Skin supported by full-depth honeycomb)	Bead-stiffened monolithic graphite/epoxy.	Bead-stiffened formed aluminum sheet.
Assembly	Upper and lower halves post-bonded with leading edge strip.	Ribs bonded to spar. Skins riveted to spar and ribs.	Ribs riveted to spar. Skins riveted to ribs and spars.
Attachment	Two hinge bolts; one actuator bolt.	Same as Concept I.	Same as Concept I.
Loads	Light except for spar caps.	Same as Concept I.	Same as Concept I.
Access	No internal access for inspection.	Inspection panels.	Inspection panels.
Source	Reference 2	Reference 1	UH-60A

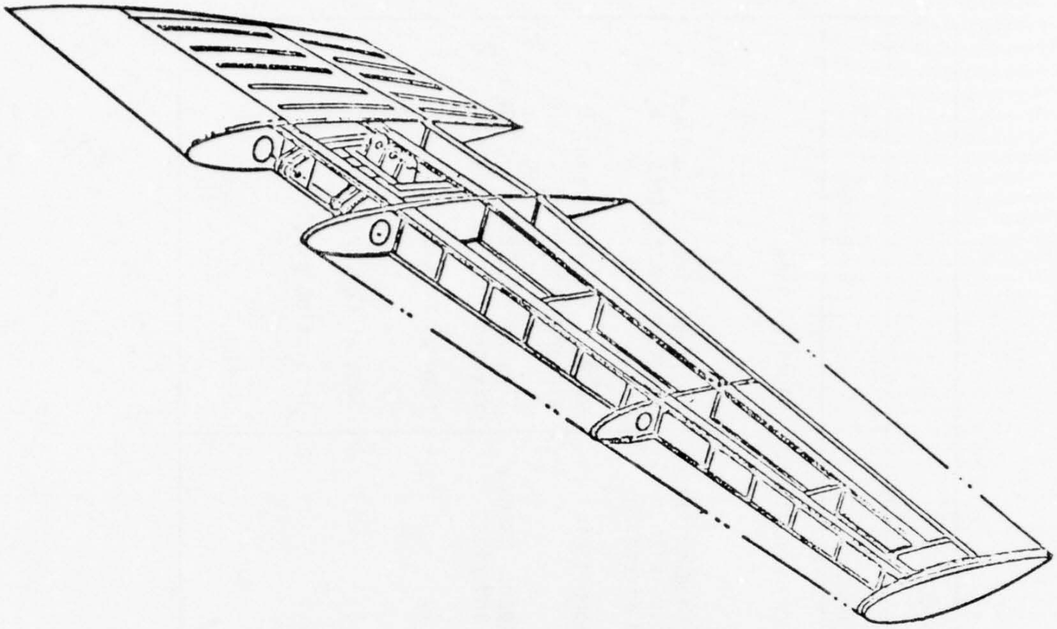


Figure 91. Composite Stabilator

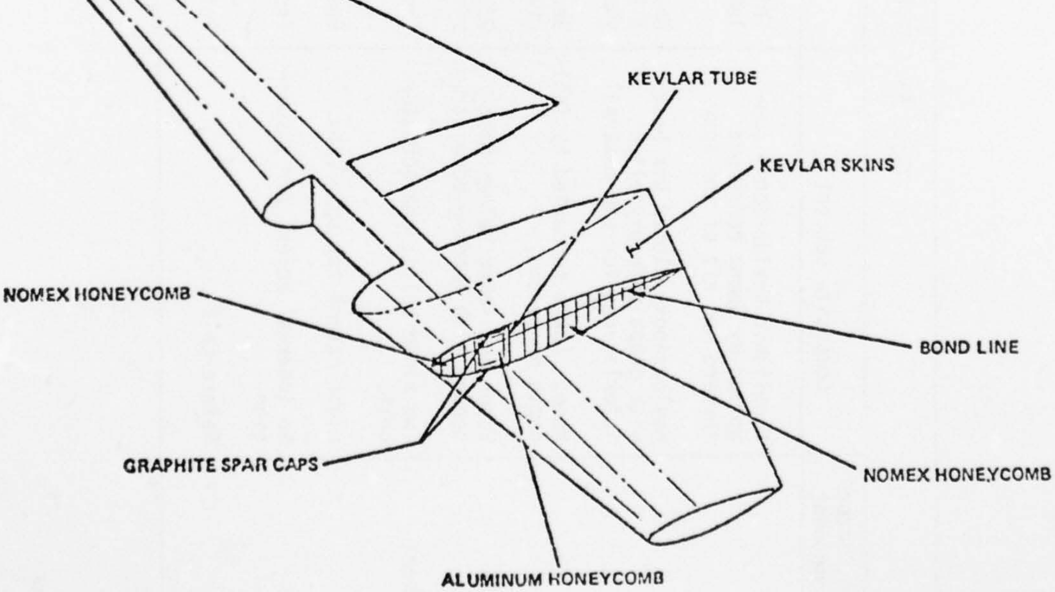


Figure 90. UH-60A Metal Stabilator

TABLE 54. REAR FUSELAGE DESIGN CONCEPTS

Design Factor/ Sub-Component	Composite Concept I	Composite Concept II	Baseline
Skin	Monolithic Kevlar/epoxy skins; 4 ply minimum, with small Nomex honeycomb panels. Titanium in upper walkway - fire zone.	Sandwich graphite/epoxy facings; aluminum honeycomb core. Minimum (2 ply) faces. Titanium walkway - fire zone.	Aluminum sheet. Titanium in fire zone.
Frames	Closed section Kevlar/epoxy with graphite/epoxy cap reinforcement.	Closed section graphite/epoxy.	Open section formed aluminum channels.
Bulkheads	Sandwich Kevlar/epoxy facings with Nomex honeycomb core.	Sandwich graphite/epoxy facings with aluminum honeycomb core.	Sandwich fiberglass/epoxy facings with aluminum honeycomb core.
Stringers and Panel Breakers	Closed section Kevlar/epoxy with graphite/epoxy cap reinforcement.	None.	Open sections formed from aluminum sheet.
Assembly	Upper and lower major assemblies each a bonded skin-skeleton. Final assembly via riveted splice at max width line.	Sandwich skin panels riveted and bonded to frames. Upper and lower assemblies same as Concept I.	Standard riveted assembly.
Attachment	Standard riveted joints along forward attachment. Bolted-on tail cone.	Same as Concept I.	Same as Concept I.

TABLE 54 (Concluded)

Design Factor/ Sub-Component	Composite Concept I	Composite Concept II	Baseline
Loads	Generally light due to large cross section. Concentrated loads at tail cone attachments.	Same as Concept I.	Same as Concept I.
Access	Interior and exterior accessible except for area under fuel cell.	Same as Concept I.	Same as Concept I.
Interfaces	Auxiliary power unit; fuel tank, fittings and lines; tail rotor drive shaft support; air retrieval fitting.	Same as Concept I.	Same as Concept I.
Source	Synthesized	Synthesized	UH-60A

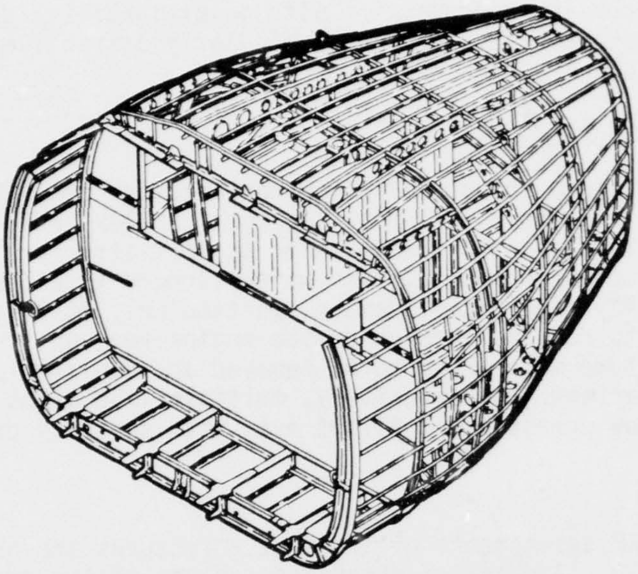


Figure 92. Metal Rear Fuselage

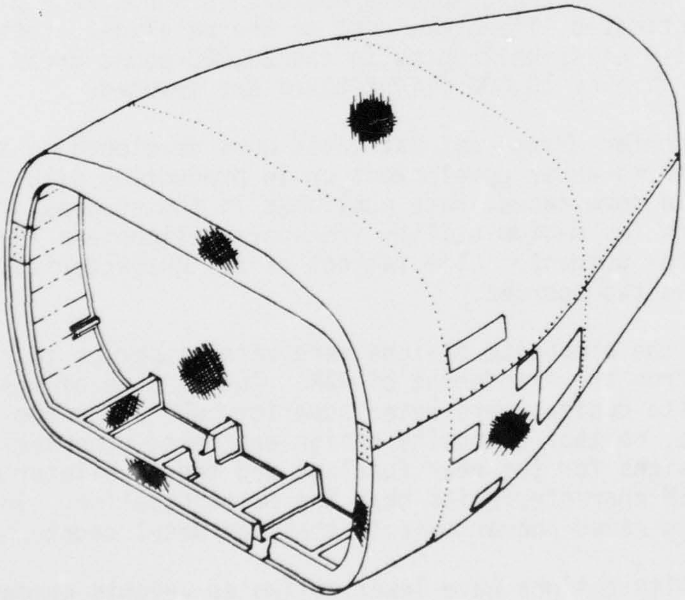


Figure 93. Composite Rear Fuselage

design concept discussed in connection with maintainability design factors earlier in the report appears to be a particularly attractive option for a composite rear fuselage design.

Transmission Support Structure

The transmission support structure design concepts are described in Table 55 and illustrated in Figures 94 and 95. The most complex and highly loaded structure in the airframe, the transmission support structure frequently employs bonded or co-cured construction which renders it an integral part of the fuselage. Any significant damage, particularly ballistic damage, could be expected to require complex custom-engineered repairs. Entire sections of the structure might have to be removed and new sections spliced in at noncritical locations. Alternatively, multiple load paths might have to be provided to allow unrestricted flight even with severely damaged members.

ANALYSIS RESULTS

Summaries of the R&M assessments of the four structures are presented in Tables 56 through 59. A detailed assessment of one of the two composite rear fuselage designs (Concept I) is given in Appendix C.

Table 60 presents the results of the cost analysis. Listed for each of the twelve designs are the estimated weight and manufacturing cost. Also shown is a qualitative ranking of the survivability/vulnerability (S/V) characteristics of the composite designs relative to the metal baselines. The estimated life-cycle cost of the composite designs is shown as a plus or minus delta from the estimated life-cycle cost of the baseline. Costs were estimated for a utility class helicopter in the 20,000-pound gross weight category. A service life of 10,000 flight-hours was assumed.

The weight and manufacturing cost estimates were developed on the basis of comparable structures under development or in production with Sikorsky helicopter models. In some cases, data published in the studies of advanced structural designs for medium utility transport helicopters (References 1 and 2) was used for guidance. The ratings of S/V characteristics were extracted from these two sources.

Overall, four of the composite designs were rated superior to the metal baseline design from the standpoint of R&M. In the case of the cockpit canopy, both composite designs were rated superior, while for the transmission support structure, neither composite design was rated superior. One of the two composite designs for the rear fuselage and the stabilator were judged to have better R&M characteristics than the metal baseline. In no case was a composite design rated poorer overall than the metal counterpart.

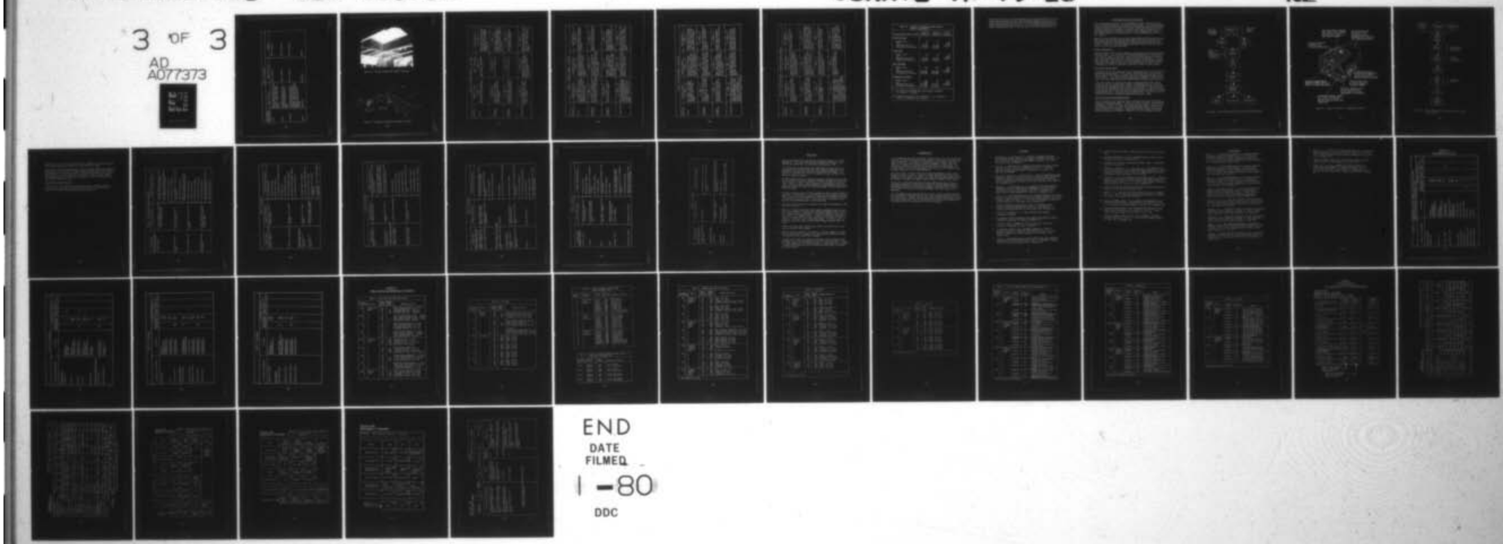
All of the composite designs have lower estimated weights compared to the metal baseline designs. Four of the eight composite designs have a lower estimated manufacturing cost. In the case of the cockpit canopy, both composite designs were judged less expensive to manufacture than a metal structure, while in the case of the transmission support structure, neither composite design was judged less expensive than the metal counterpart.

TABLE 55. TRANSMISSION SUPPORT STRUCTURE DESIGN CONCEPTS

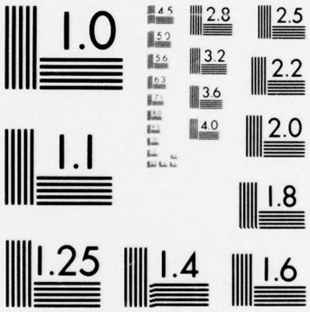
Design Factor/ Sub-Component	Composite Concept I	Composite Concept II	Baseline
Skin	Monolithic Kevlar/epoxy with interlayered graphite/epoxy for reinforcement. .08-.10 inch thick.	Sandwich; Kevlar/epoxy facings with Nomex core. Post-bonded to frame members. 4 ply faces.	Aluminum sheet.
Frames and Beams	Open-section I-beams, monolithic graphite/epoxy.	Sandwich, graphite/epoxy facings with Nomex core. Co-cured assemblies.	Open-section aluminum I-beams built up from sheet and extrusions. Riveted assemblies.
Stiffeners	Foam filled hat sections.	None.	Open section channels made from formed aluminum sheet.
Fittings	No separate fittings for transmission attachment. Graphite/epoxy plies added to basic frames as required.	Machined aluminum fittings bolted to basic frame structures for transmission attachment.	Integral mounting points machined into frame center sections for transmission attachment.
Assembly	Framing members and skin co-cured together.	Framing members joined with bonded pultrusion angles. Transmission mount fittings bolted to corners of main frame-beam intersections.	Basic framing joined with lock bolts. Smaller framing members and skin riveted to basic framing.
Attachment	Structure joined to forward and aft cabin roofs with lockbolts and rivets and to door interface structure along sides with rivets.	Same as Concept I.	Same as Concept I.

AD-A077 373 UNITED TECHNOLOGIES CORP STRATFORD CT SIKORSKY AIRCR--ETC F/G 15/5
ADVANCED STRUCTURES CONCEPTS R AND M/COST ASSESSMENTS.(U)
SEP 79 T N COOK , B F KAY DAAJ02-77-C-0061
UNCLASSIFIED SER-510016 USARTL-TR-79-18 NL

3 OF 3
AD
A077373



END
DATE
FILMED
1-80
DDC



MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS-1963-A

TABLE 55 (Concluded)

Design Factor/ Sub-Component	Composite Concept I	Composite Concept II	Baseline
Loads	Main frames and beams are most heavily loaded structure in the aircraft. High concentrated loads at transmission attach points.	Same as Concept I.	Same as Concept I.
Access	Equipment mounted above and below the structure restricts access for inspection and repair.	Same as Concept I.	Same as Concept I.
Interfaces	Transmission mounting points; controls mounting brackets; electronic boxes and mounting brackets; electrical wiring and hydraulic lines; cabin soundproofing; passenger seat mounting.	Same as Concept I.	Same as Concept I.
Source	Reference 3	Reference 1	UH-60A

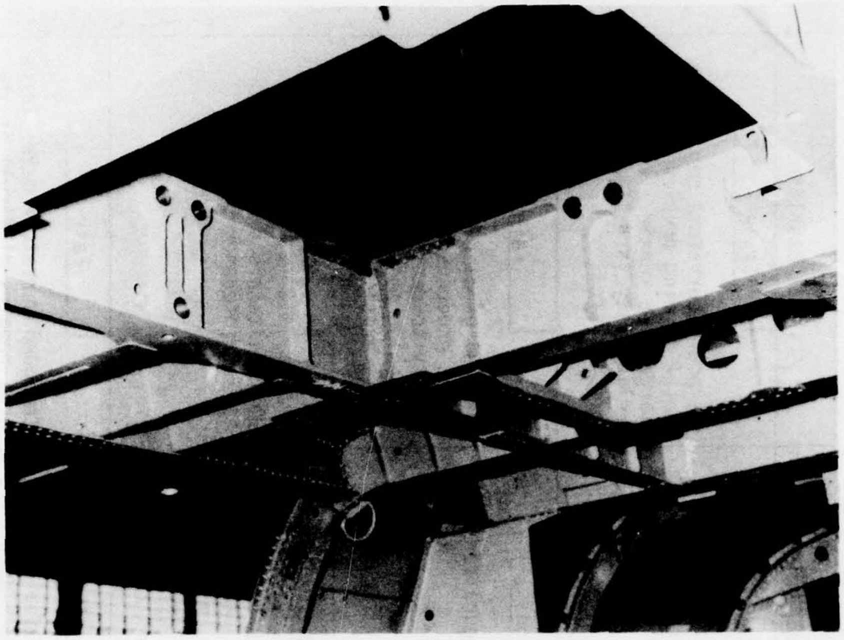


Figure 94. UH-60A Transmission Support Structure

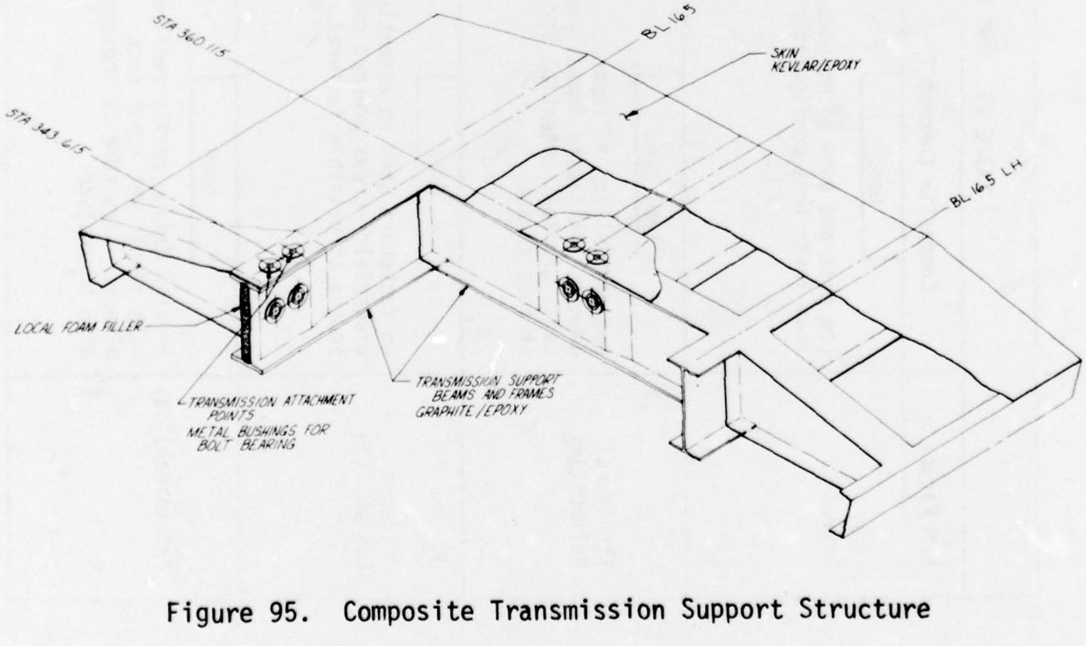


Figure 95. Composite Transmission Support Structure

TABLE 56. R&M RATING SUMMARY - COCKPIT CANOPY

R&M Factor	Composite Concept I	Composite Concept II	Metal Baseline
Overall	Good Low cost and ease of maintenance favor this configuration.	Good Higher cost of replacement than Configuration I.	Fair Compound curvature may create metal forming problems for large area repair. Hardware reliability is a potential problem area.
Structural Reliability	Very Good Low probability of impact damage. Ballistic impact presents greatest potential for serious damage.	Very Good Similar to Configuration I.	Fair Minimal gages of aluminum sheet produces high susceptibility to denting and buckling. Corrosion possible if water traps exist.
Hardware Reliability	Good No fasteners used in assembly. Windshield screws present only source of possible failures.	Good Similar to Configuration I.	Poor Many fasteners create potential maintenance problem.
Maintainability	Good Routine damage easily repaired with simple wet layup techniques. Structure is inexpensive to replace.	Good Repairability equivalent to Configuration I. Higher cost materials would make replacement more expensive than Configuration I.	Fair Compound curvature makes large area repairs difficult in the field. Access restrictions require use of blind fastening methods.

TABLE 57. R&M RATING SUMMARY - STABILATOR

R&M Factor	Composite Concept I	Composite Concept II	Metal Baseline
Overall	Good Full-depth honeycomb construction avoids need for access to interior of unit. Small damage repairable with established plug/patch techniques.	Fair Damage affecting sub-structure requires disassembly for repair. Graphite splintering could seriously affect structural integrity.	Fair Damage affecting sub-structure requires partial disassembly for repair.
Structural Reliability	Good No corrosion. Cracking unlikely with full-depth honeycomb. Small punctures and local crushing at edges most likely damage.	Poor No corrosion. Thin laminates allow serious puncturing, buckling, crushing and cracking from impact. Graphite splintering spreads damage.	Fair Denting most probable type of damage. Corrosion may develop.
Hardware Reliability	Very Good No access panels. Mounting bolts are only fasteners.	Fair Potential 1/4 turn fastener problems. Access panel removal and installation may damage fastener holes. Assembly rivets may corrode and/or work loose.	Poor Larger number of fasteners. Potential 1/4 turn fastener problems at access panels. Assembly rivets may corrode and loosen.
Maintainability	Good Small dents and punctures easily repaired with established fiberglass techniques except for spar/web area which is buried in honeycomb. Entire assembly replaceable with 3 bolts.	Fair Bond repairs only. No wet lay-up method qualified for graphite repairs. Difficult access for substructure damage repair.	Good Sheet metal repairs well established, but parts forming difficult in field. Limited access for substructure damage repair.

TABLE 58. R&M RATING SUMMARY - REAR FUSELAGE

R&M Factor	Composite Concept I	Composite Concept II	Metal Baseline
Overall	Good No corrosion. Few fasteners. Small repairs easily made. Development of quick-fix and large area repair methods needed.	Fair Repair more difficult than Configuration I. Frequent denting, puncturing and crushing could create maintenance problem. Core corrosion could affect structural integrity.	Fair Hardware reliability is most serious concern. Corrosion is potential problem in tub area. Repairs difficult if metal forming is required.
Structural Reliability	Good Majority of small impact damage is non-critical. No corrosion. Stowage area may be susceptible to damage.	Fair Small impact loads could cause splintering of thin graphite. Aluminum core subject to corrosion if moisture leaks through faces.	Good Some corrosion problems likely in tub area. High frequency of denting, but primarily cosmetic.
Hardware Reliability	Good Bonded structure eliminates most fasteners. Some failures possible at access panels.	Fair Fewer rivets than aluminum; most are highly corrosion resistant due to graphite compatibility. Some hardware problems at access panels.	Poor Corrosion and loosening of rivets potential problem. Could be severe in tub section.
Maintainability	Good Small repairs easily done, but resin cure times are currently excessive without heat. New resins offer potential for improvement. Consideration should be given to modular construction for large area repairs.	Fair Graphite repair is difficult. No qualified methods currently available for repair with composites; metal patches must be used.	Good Established techniques for small area repairs. Metal forming may be difficult for large area repairs in field.

TABLE 59. R&M RATING SUMMARY - TRANSMISSION SUPPORT STRUCTURE

R&M Factor	Composite Concept I	Composite Concept II	Metal Baseline
Overall	Fair High cost of basic raw materials and difficulty of graphite repair are most serious problems.	Fair Sandwich panels are more difficult to repair than monolithic. Aluminum fittings are replaceable, but could be corrosion problem with graphite.	Fair Machined fittings difficult to repair or replace.
Structural Reliability	Fair Skin is most susceptible to damage. Abrasion likely; could be serious due to high loading. Ballistic damage unlikely due to location.	Fair Similar to Configuration I. Puncturing and crushing more likely due to more rigid panels and thin faces. Corrosion possible with aluminum against graphite.	Fair Significant denting and buckling may occur due to dropped tools and foot traffic. Abrasive removal of corrosion protection could be problem for the skin.
Hardware Reliability	Good Very few fasteners used. Installation fasteners are only potential problem.	Good Lockbolts used to attach transmission mount fittings to composite structure could be maintenance problem.	Poor Conventional aluminum riveted construction. High loads produce fastener problems.
Maintainability	Poor Limited access makes repair difficult. Graphite difficult to patch. Monolithic Kevlar skins can be patched using fiberglass techniques. Structure is expensive and difficult to replace. However, extensive damage is unlikely.	Poor Similar to Configuration I. Sandwich panels somewhat more difficult to repair than monolithic.	Poor Machined fittings are difficult or impossible to patch. Lighter frame members can be patched, but forming is difficult in the field. Extensive damage unlikely.

TABLE 60. SUMMARY OF ADVANCED DESIGN CONCEPTS R&M/COST ASSESSMENTS

Structure/Evaluation Factor	Composite Concept I	Composite Concept II	Metal Baseline
Cockpit Canopy			
Weight (lbs.)	47	35	49
S/V*	+	+	Base
Manufacturing Cost	$4,600	$5,100	$5,200
Life-Cycle Cost Delta**	-$3,232	-$2,620	--
Stabilator			
Weight (lbs.)	62	56	68
S/V	+	+	Base
Manufacturing Cost	$5,800	$6,600	$6,200
Life-Cycle Cost Delta	-$4,040	+$3,620	--
Rear Fuselage			
Weight (lbs.)	380	359	422
S/V	+	-	Base
Manufacturing Cost	$29,000	$55,000	$47,000
Life-Cycle Cost Delta	-$19,356	+$12,248	--
Transmission Support			
Weight (lbs.)	88	83	110
S/V	Same	+	Base
Manufacturing Cost	$18,000	$19,500	$16,500
Life-Cycle Cost Delta	+$ 1,490	+$ 3,010	--

* Survivability/Vulnerability rating versus baseline (+ = better; - = poorer)

** Estimated life-cycle cost reduction (-) or increase (+) relative to baseline, per aircraft

Predicted life-cycle costs show reductions over the metal baseline costs for those composite designs having lower estimated manufacturing costs and increases over the metal baseline costs for those composite designs having higher estimated manufacturing costs. This illustrates the dominant influence of manufacturing cost in the life-cycle cost of airframe structures.

PRELIMINARY R&M DESIGN GUIDELINES

This program represents the first comprehensive study of the R&M implications of advanced composite structures for helicopters. While new knowledge has been gained from this work and many R&M issues have been placed in perspective, further work remains to be done in the areas of damage tolerance and damage mitigation techniques and field-level inspection and repair. These requirements are discussed further in the Recommendations section of this report.

When this additional R&D work has been accomplished and more actual service experience with advanced composites has been acquired, the publication of a formal R&M guide for advanced composite structures design should be considered. Meanwhile, some preliminary guidelines evolving from the current state of the art can be established.

FLOW OF R&M ACTIVITY

Advanced composite structures represent a relatively new field of technology for helicopters. Less is known of the R&M characteristics of these structures than is known of most other areas of the aircraft where the technology, while continually evolving, is relatively well-established. This suggests that R&M in the design of advanced structures be approached somewhat differently from general practices in R&M engineering, at least until the technology and level of experience reach that of other aircraft systems.

Preliminary Design Phase

Figure 96 outlines the suggested flow of R&M engineering activity during the preliminary design phase. At the concept evaluation stage, an R&M assessment is made of each design candidate using the technique outlined earlier in this report. This yields a gross comparative assessment of R&M and life-cycle costs and highlights potential R&M problems related to each concept.

Trade-off studies are conducted and R&M is weighed against other design attributes such as producibility, weight and manufacturing cost to arrive at the optimum cost/performance solution. Design options identified by the trade-offs are synthesized into a final design concept. At the conclusion of this phase, R&M criteria for detailed design are established in the areas of damage tolerance, serviceability and repairability. Figure 97 is an example of the type of information generated from such a study.

Detailed Design and Development Phase

Figure 98 outlines the suggested flow of R&M activity during the detailed design and development phase. Prior to the start of design, a detailed R&M assessment is made of the final preliminary design concept. Areas of the design appearing not to satify the established R&M criteria are discussed and resolved with the designer. Where published data leaves uncertainty about the probable performance of materials in service (damage tolerance,

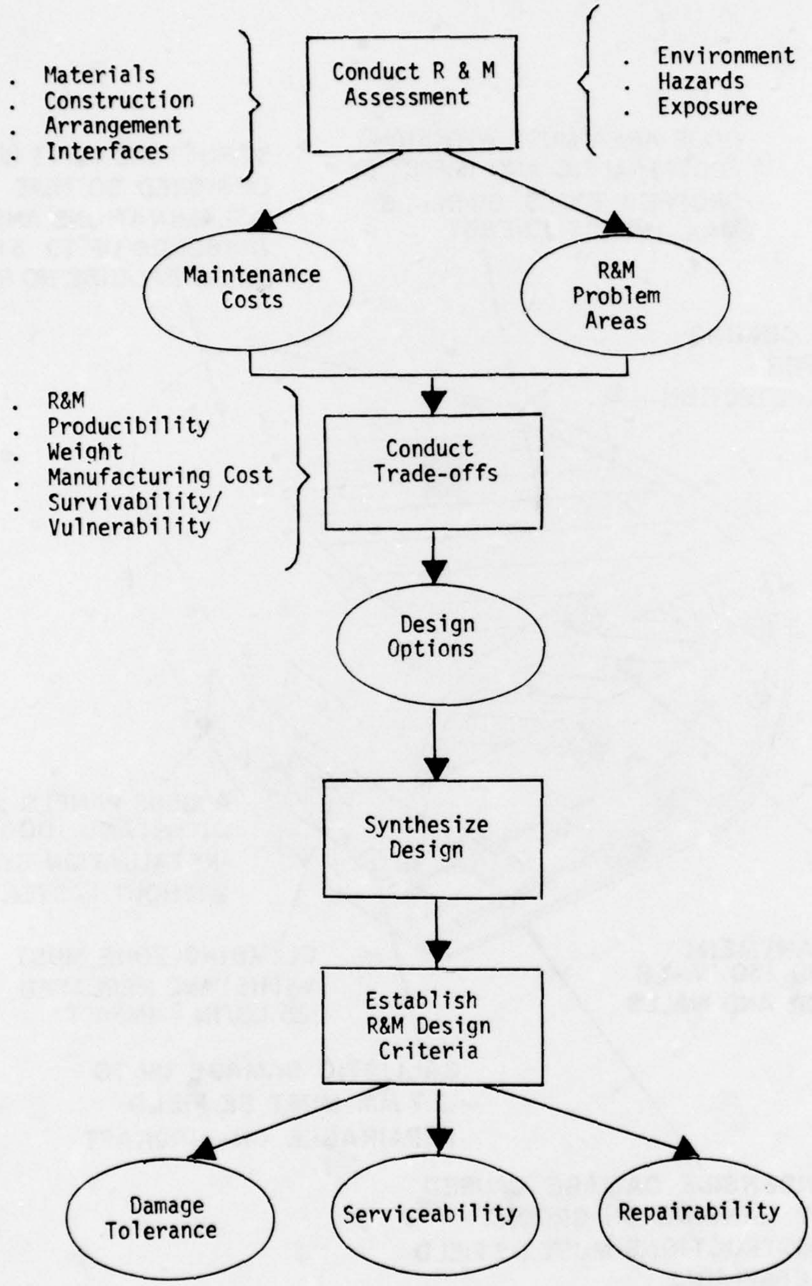


Figure 96. Flow of R&M Activities During Preliminary Design

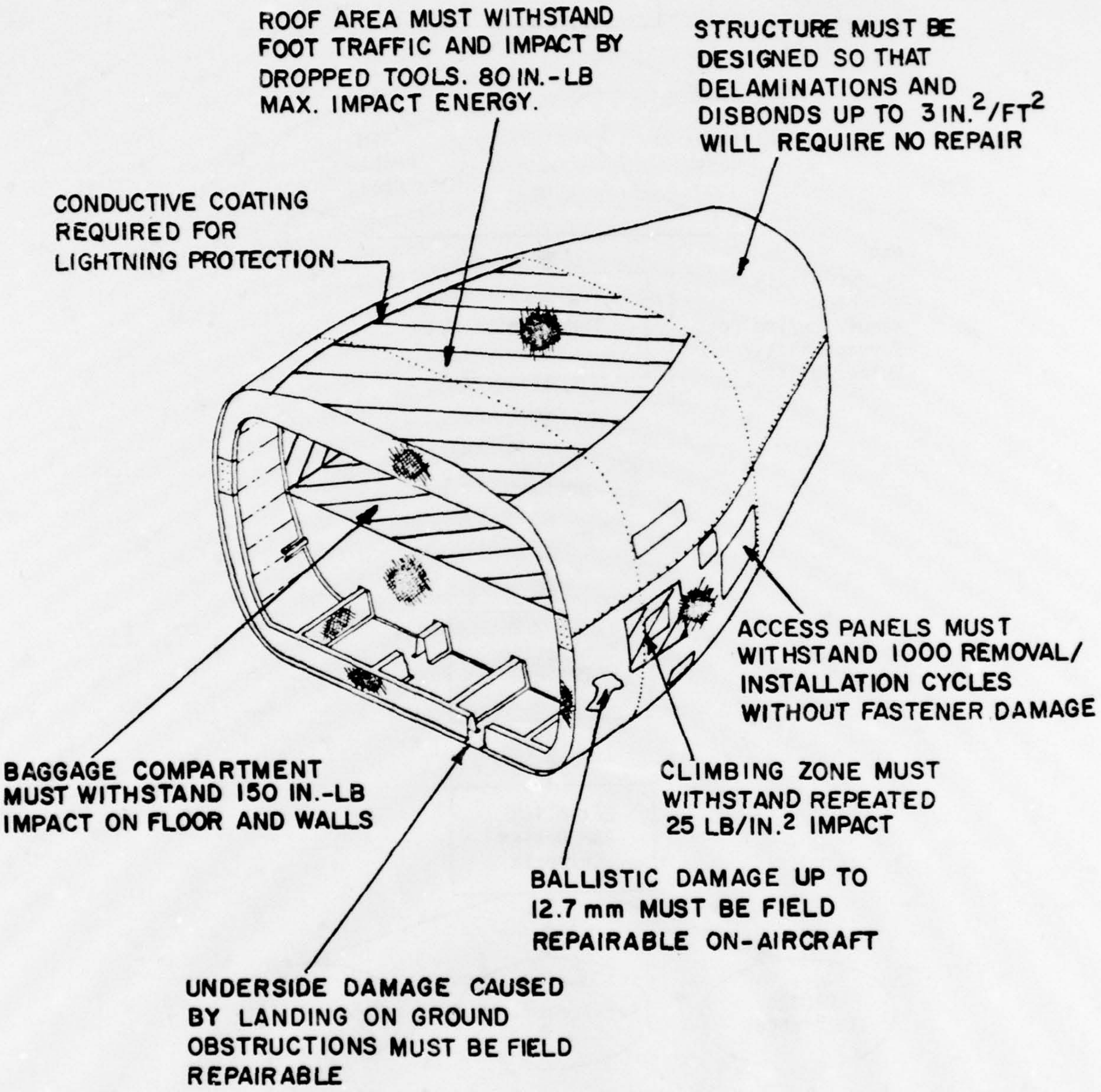


Figure 97. Typical Illustration of R&M Design Criteria

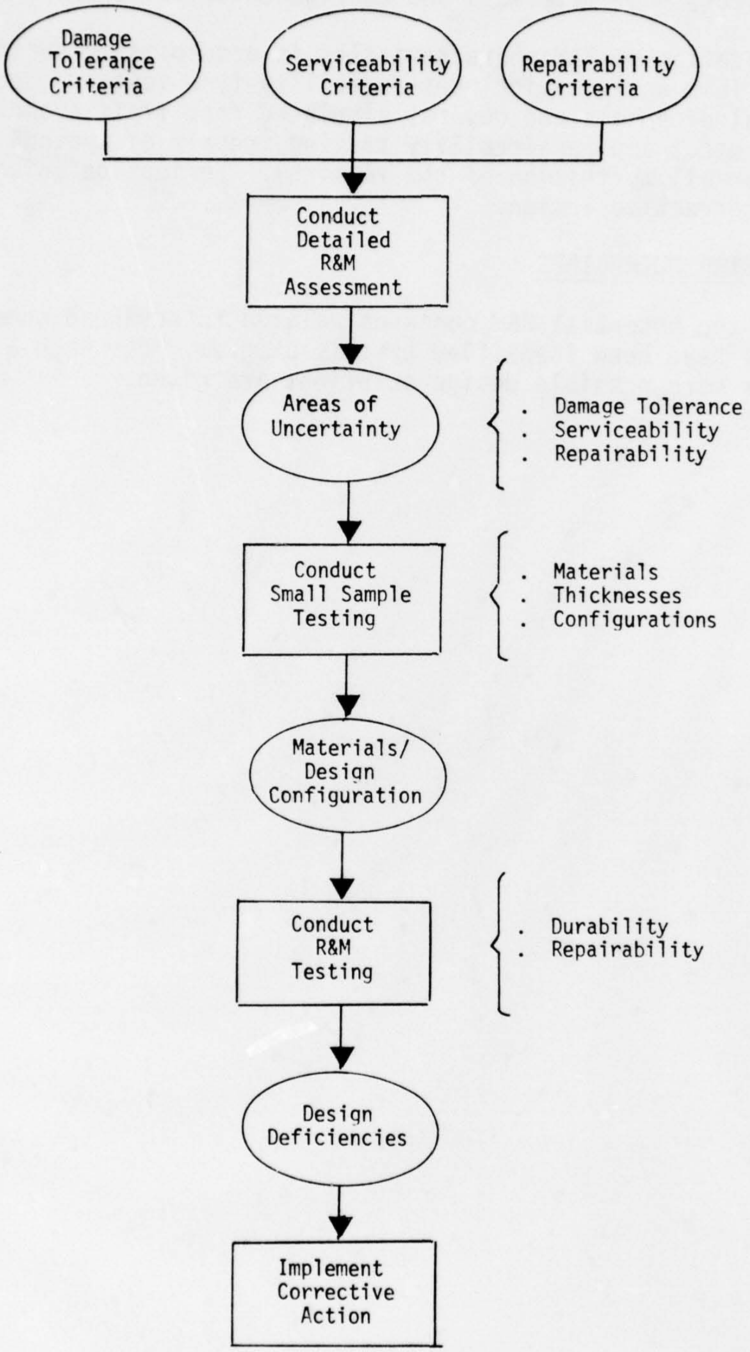


Figure 98. Flow of R&M Activities During Detailed Design and Development

repairability, etc.) small sample testing is conducted to resolve the issue, and a final selection of materials and configuration is made.

Functional evaluation of R&M characteristics is accomplished during development testing. This activity includes durability-type testing (door slamming, repeated removal of panels and covers, simulated foot traffic and foreign object impact, etc.) and repairability testing (repair of typical damage and strength/durability testing of the repairs). Serious deficiencies are scheduled for corrective action.

PRELIMINARY DESIGN GUIDELINES

Table 61 lists the potential R&M concerns related to advanced composite structures that have been identified by this program. For each area of concern, one or more possible design solutions are given.

TABLE 61. PRELIMINARY R&M DESIGN GUIDELINES

Design Factor	Potential R&M Concern	Potential Design Solution
Construction-Related		
Sandwich panels	● Loss of compression stability (buckling failures) due to light to moderate impact.	● Increase moment of inertia to reduce sensitivity to impact defects. ● Use impact resistant facing material or increase facing thickness. ● Add protective coverings. ● Use monolithic construction.
Aluminum honeycomb panels	● Moisture entry around inserts and corrosion.	● Use non-metallic or corrosion resistant materials. ● Add sealant around holes. ● Replace inserts with anchor nuts attached to bonded zee closures.
Nomex honeycomb panels	● Broken/crushed core due to impact.	● Increase panel thickness. ● Increase core density. ● Increase facing thickness. ● Use monolithic construction.
Composite-faced honeycomb panels	● Surface fractures and penetrations due to light to moderate impact.	● Increase facing thickness. ● Increase panel flexibility (energy absorption capabilities). ● Add protective covering.

TABLE 61 (Continued)

Design Factor	Potential R&M Concern	Potential Design Solution
Aluminum-faced/aluminum honeycomb panels	● Denting due to light to moderate impact.	● Use rigidized skins. ● Increase skin gage. ● Increase core density. ● Use unidirectional fiberglass facings. ● Use Nomex core. ● Add protective covering.
Stiffeners, open-section	● Twisting, buckling type failures.	● Provide intermediate supports. ● Substitute closed-section members. ● Use sandwich construction.
<u>Gage/Thickness Related</u>		
Thin monolithic composites	● Punctures at low energy levels.	● Increase material thickness. ● Use more impact-resistant materials.
Thick monolithic composites	● Subsurface damage (delamination) due to heavy impact.	● Provide sacrificial facing. ● Provide multiple load paths - redundancy. ● Limit stress levels to impede damage propagation.

TABLE 61 (Continued)

Design Factor	Potential R&M Concern	Potential Design Solution
Minimum gage construction	● Damage susceptibility.	● Avoid composite materials less than .020 inch thick. ● Alter structural geometry to enable use of thicker materials without compromising structural efficiency.
Thin unidirectional graphite/epoxy	● Face splintering due to impact.	● Use woven graphite fabric for outermost ply on exposed surfaces.
Fastener/Attachment Related		
Mechanically fastened composite structure	● Fastener overtorque damage.	● Provide grommets. ● Use large head fasteners. ● Avoid flush head fasteners. ● Provide adequate thickness.
Mechanically fastened composite structure	● Fastener hole elongation and tearout.	● Provide adequate reinforcement at holes. ● Use large diameter fasteners. ● Provide large edge distance. ● Insert metal shims in laminate.

TABLE 61 (Continued)

Design Factor	Potential R&M Concern	Potential Design Solution
Load Intensity Related		
Heavily loaded tension members (longerons, etc.)	● Loss of strength due to moderate impact damage.	● Provide sufficient strength to withstand normal service damage.
Heavily loaded structure	● Damage propagation (cracks, delamination).	● Use increased damage tolerance materials.
		● Use crack arrestors.
		● Design for low stress levels.
Material Compatibility Related		
Graphite-to-metal interface	● Corrosion.	● Avoid condition.
		● Provide insulation at faying surfaces.
Interfaces Related		
Equipment mounts and supports	● Loosening, fretting and fastener hole wear.	● Reinforce mounting areas.
Edges of aluminum honeycomb panels	● Moisture entry and corrosion.	● Provide sealants to prevent moisture accumulation.
		● Use non-wicking adhesives.
		● Use corrosion resistant materials.
Cutouts	● Edge delamination.	● Increase thickness.
		● Add protective doublers.
		● Avoid conditions where thin edges are directly exposed.

TABLE 61 (Continued)

Design Factor	Potential R&M Concern	Potential Design Solution
Component Related		
Fairing, cowling	● Edge and corner damage due to rough handling.	● Increase thickness. ● Add protective doublers.
Fairings	● Abrasion/fretting damage in high vibratory environment.	● Reinforce wear points (attachment surfaces). ● Add protective doublers. ● Increase thickness.
Doors	● Buckling, warping, misalignment due to slamming.	● Provide sufficient strength to withstand loading. ● Verify with testing.
Doors	● Hinge-to-structure damage due to slamming open.	● Same as above.
Walkways, floors	● Abrasion damage.	● Provide wear resistant surface coating (polyurethane paint). ● Provide sacrificial or protective covering. ● Substitute metal facings.
Joints and fittings	● Damage due to impact.	● Provide sufficient margin to withstand normal service abuse.

TABLE 61 (Concluded)

Design Factor	Potential R&M Concern	Potential Design Solution
Maintenance Related		
Co-cured and adhesively bonded joints	● Disassembly.	● Provide means for in-place repair. ● Use modular design concept (repair strips).
Environment Related		
Composites	● Environmental deterioration (strength, stiffness).	● Apply appropriate strength reduction factors during structural design and qualification.
Composites	● Fatigue damage.	● Same as above.
Composites	● Burns, delamination due to lightning strikes.	● Provide conductive covering (flame spray, aluminum tape or wire mesh).

CONCLUSIONS

1. Service experience with conventional airframe structures is not well documented. Experience with advanced composite structures is very limited and useful only for qualitative assessments of R&M.

2. The limited service experience with composites plus the results of documented test programs have shown that advanced composites can be designed to withstand the effects of the natural environment. The majority of failures will occur from external causes, primarily as a result of damage by impact.

3. The R&M characteristics of advanced structures concepts can be assessed qualitatively through a systematic analysis of material factors and design factors. Areas of design concern are highlighted in the process. Life-cycle costs can be estimated by comparing attributes of an advanced concept with those of a baseline design for which equivalent costs are known.

4. Airframe structures require minimal upkeep and repair during peacetime operation of helicopters. The airframe has a high frequency of maintenance, but the cost of maintenance is relatively low compared with other subsystems of the aircraft. Life-cycle cost is dominated in most cases by the manufactured cost of the structure.

5. Composites appear to have the potential for modest improvements in R&M and life-cycle cost.

6. Based on the monolithic panel impact testing conducted under this program, for equivalent thicknesses of material,fiberglass rates as the most damage tolerant, graphite the second most damage tolerant and Kevlar slightly less damage tolerant. Sandwich construction has poorer impact resistance than monolithic construction and tends to suffer reductions in strength due to subsurface damage. Of the two types of honeycomb, Nomex was shown to have better damage tolerance than aluminum.

7. Simple field-type repair methods were shown to be effective for many types of routine impact damage.

8. Based on the test results, a thickness of .020 inch appears to represent a minimum gage for composites in applications where minor impact is expected at some significant frequency.

9. Inspection and repair are among the important issues confronting the introduction of advanced composite structures for helicopters. Repair of combat damage is of particular concern. Further work will be needed to develop techniques that are cost-effective and suited to the Army field environment.

RECOMMENDATIONS

It is recommended that the Army conduct further work in the area of advanced composite structures R&M specifically related to field-level inspection and repair techniques. Two aspects of advanced composites repair have been shown by this program to require further development. Simple, fast-cure (quick-fix) repairs are required for routine service-related damage. Improved repairs of this type are needed, particularly for the advanced high-strength composites such as graphite/epoxy.

The second aspect of repair requiring further development is that of major damage to primary structure. Repair of combat damage presents particular problems in this regard. Techniques for repairing large area damage should be explored, including the modular design concept proposed in this report.

Future work in advanced composite structures R&M should also include a thorough examination of such related factors as improved damage tolerance and damage mitigation techniques, and component serviceability criteria. Field inspection methods and damage assessment techniques should also be addressed.

It is recommended that the continuation of work in this area be associated with the design and development of a full-scale advanced composite structure for a helicopter. Ideally, the structure will be relatively complex so that the range of R&M options can be explored and the results can be made generically applicable to a spectrum of aircraft structures.

REFERENCES

1. Hoffstedt, P. J.,and Swatton, S., ADVANCED HELICOPTER STRUCTURAL DESIGN INVESTIGATION, Boeing Vertol Company, USAAMRDL-TR-75-56A, U. S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Va., March 1976, AD A0224662.

2. Rich, M. J., INVESTIGATION OF ADVANCED HELICOPTER STRUCTURAL DESIGNS, Sikorsky Aircraft Division, USAAMRDL-TR-75-59A, U. S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Va., May 1976, AD A026246.

3. Kay, B. F., Lowry, D. S., and Rich, M. J., STUDY TO INVESTIGATE DESIGN, FABRICATION AND TEST OF LOW COST CONCEPTS FOR LARGE HYBRID HELICOPTER FUSELAGE - PHASE III, Sikorsky Aircraft Division, NASA Contractor Report 158988, National Aeronautics and Space Administration, Hampton, Va., February 1979.

4. Barrett, L. D., and Aronson, R. B., RELIABILITY AND MAINTAINABILITY PROGRAM FOR SELECTED SUBSYSTEMS AND COMPONENTS OF CH-47 and UH-1 HELICOPTERS, Boeing Vertol Company, Report No. D210-10846-1, U. S. Army Aviation Systems Command, St. Louis, Mo., September 1974.

5. Labor, J. D., SERVICE/MAINTAINABILITY OF ADVANCED COMPOSITE STRUCTURES, Quarterly Progress Report No. 4, Northrop Corporation Report No. 77-157, Contract F33615-76-C-3142, U. S. Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio, November 1977.

6. ADVANCED COMPOSITES DESIGN GUIDE, VOLUME IV, MATERIALS, Third Edition, Advanced Development Division, Air Force Materials Laboratory, Wright-Patterson Air Force Base, Ohio, January 1973.

7. KEVLAR 49 DATA MANUAL, E. I. DuPont DeNemours and Company, Wilmington, Delaware.

8. MIL-HDBK-5B, METALLIC MATERIALS AND ELEMENTS FOR AEROSPACE VEHICLE STRUCTURES, Department of Defense, September 1971.

9. SCOTCHPLY PRODUCT INFORMATION, SP-114, Industrial Specialties Division, 3-M Company, St. Paul, Minnesota.

10. FLIGHTWORTHY GRAPHITE FIBER REINFORCED COMPOSITES, VOLUME 1, Northrop Corporation, Report No. AFML-TR-20-207, U.S. Air Force Materials Laboratory, Wright-Patterson Air Force Base, Ohio, October 1970.

11. Flonc, N., CHARACTERIZATION OF BORON, GRAPHITE AND GLASS FILAMENT/ ORGANIC MATRIX COMPOSITE MATERIALS, Report No. SER-50644, Sikorsky Aircraft Division, Stratford, Conn., January 1970.

12. SIKORSKY STRUCTURES MANUAL, Sikorsky Aircraft Division, Stratford, Conn.

13. MATERIAL PROPERTIES OF HEXCEL HONEYCOMB MATERIALS, TSB120, Hexcel Corporation, Dublin, California, 1975.

14. MIL-HDBK-17A, PLASTICS FOR AEROSPACE VEHICLES, PART I, REINFORCED PLASTICS, January 1971.

15. Foreman, C., McGovern, S. A., and Knight, R., S-34 GRAPHITE/EPOXY SPOILER FABRICATION OF TEN SHIPSETS AND DAMAGE REPAIR STUDY, Vought Corporation Systems Division, Report No. NADC-76234-30, Naval Air Development Center, Warminster, Pa., May 1976.

16. La Salle, R. M., REPAIR PROCEDURES FOR ADVANCED COMPOSITE STRUCTURES, VOLUME II, REPAIR GUIDE, General Dynamics Corporation, Report No. AFFDL-TR-76-57 Volume 2, U. S. Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio, December 1976.

17. FLEXURE TEST OF FLAT SANDWICH CONSTRUCTIONS, American Standard Test Method No. ASTM C393-62, Reapproved 1970.

18. Reddick, H. K., ARMY HELICOPTER COST DRIVERS, Report No. USAAMRDL-TM-7, U. S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Va., August 1975, AD 015517.

19. EXECUTIVE SUMMARY REPORT, CH-47 ASSESSMENT AND COMPARATIVE FLEET EVALUATIONS, FINAL REPORT, USAAVSCOM Technical Report No. 74-46, U. S. Army Aviation Systems Command, St. Louis, Mo., November 1974.

20. EXECUTIVE SUMMARY REPORT, UH-1H ASSESSMENT AND COMPARATIVE FLEET EVALUATIONS, USAAVSCOM Technical Report No. 75-3, U. S. Army Aviation Systems Command, St. Louis, Mo., April 1975.

21. EXECUTIVE SUMMARY REPORT, OH-58A FLEET ASSESSMENT, USAAVSCOM Technical Report No. 75-34, U. S. Army Aviation Systems Command, St. Louis, Mo., September 1975.

BIBLIOGRAPHY

1. Labor, J. D., SERVICE/MAINTAINABILITY OF ADVANCED COMPOSITE STRUCTURES, Quarterly Progress Report No. 1, Northrop Corporation Report No. 77-44, Contract F33615-76-C-3142, U. S. Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio, February 1977.

2. Labor, J. D., SERVICE/MAINTAINABILITY OF ADVANCED COMPOSITE STRUCTURES, Quarterly Progress Report No. 2, Northrop Corporation Report No. 77-94, Contract F33615-76-C-3142, U. S. Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio, May 1977.

3. Labor, J. D., SERVICE/MAINTAINABILITY OF ADVANCED COMPOSITE STRUCTURES, Quarterly Progress Report No. 3, Northrop Corporation Report No. 77-119, Contract F33615-76-C-3142, U. S. Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio, November 1977.

4. Studer, V. J., and La Salle, R. M., REPAIR PROCEDURES FOR ADVANCED COMPOSITE STRUCTURES, VOLUME I GENERAL STUDY, General Dynamics Corporation, Report No. AFFDL-TR-76-57, Volume 1, U. S. Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio, December 1976.

5. Scow, A. L., et. al., LARGE AREA COMPOSITE STRUCTURE REPAIR, Northrop Corporation Aircraft Group, Report No. AFFDL-TR-77-5, U. S. Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio, May 1977.

6. Stoecklin, R. L., 737 GRAPHITE COMPOSITE SPOILER FLIGHT SERVICE EVALUATION, Boeing Commercial Airplane Co., NASA CR-145207, Langley Research Center, NASA, Hampton, Va., August 1977.

7. Stoecklin, R. L., 737 GRAPHITE COMPOSITE SPOILER FLIGHT SERVICE EVALUATION, Boeing Commercial Airplane Co., NASA CR-144984, Langley Research Center, NASA, Hampton, Va., May 1976.

8. Wooley, J. H., FLIGHT SERVICE EVALUATION OF PRD-49/EPOXY COMPOSITE PANELS IN WIDE-BODIED COMMERCIAL TRANSPORT AIRCRAFT, Lockheed California Co., LR-26580, Langley Research Center, NASA, Hampton, Va., July 1974.

9. Putinta, M., CH-54B HELICOPTER INSPECTION REPORT, BORON/EPOXY REINFORCED STRINGERS AND TAIL SKID SUPPORT TUBES, Report No. SSD64A11575.A16, Sikorsky Aircraft Division, Stratford, CT, October 1977.

10. Watson, J. C., EFFECTS OF IN-SERVICE ENVIRONMENT OF F-4 BORON/EPOXY RUDDER, McDonnell Aircraft Co., Report No. AFFDL-TR-74-103, Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio, September 1974.

11. COMPOSITE AIRCRAFT STRUCTURES FLIGHT SERVICE COLLOQUIUM, NASA Langley Research Center, Hampton, Va., June 1974.

12. Howell, W. E., et. al., PROGRAM FOR ESTABLISHING LONG-TERM FLIGHT SERVICE PERFORMANCE OF COMPOSITE MATERIALS IN THE CENTER WING STRUCTURE OF C-130 AIRCRAFT, Lockheed Georgia Co., Contract No. NASI-11100, NASA, Langley Research Center, Hampton, Va.

APPENDIX A

ENVIRONMENTAL EFFECTS DATA

TABLE A-1. ENVIRONMENTALLY RELATED DAMAGE TO HELICOPTER STRUCTURES EXTRACTED FROM ROCKWELL INTERNATIONAL SURVEY

Structures Class/Component	Environment	Events/10^5 Flight Hours				
		Surface Damage	Strctr'l Damage	Strctr'l Deform.	Strctr'l Deter.	Total
Primary Structure						
Exterior Skin	Vibration		300			
	Fluctuating Loads		805			
	Rotor Downwash		300			
	Maintenance/Handling		340			
Interior Skin	Vibration		300			
	Rotor Downwash		300			
	Maintenance/Handling		240			
Structural Member	Fluctuating Loads		3432			
	Cleaning Fluids				150	
Honeycomb Structure	Moisture/Precipitation		583			
	Vibration		583			
	Mechanical Shock		641			
	Aircraft Fluids		583			
Cabin Structure	Fluctuating Loads		2496			
Tail Boom Structure	Fluctuating Loads		170			
Ramp Structure	Maintenance/Handling	3	14			
Engine Mount Structure	Vibration	50	50			
Vertical Stabilizer	Vibration		32			
Horizontal Stabilizer	Vibration	4	45			

TABLE A-1 (Continued)

Structures Class/Component	Environment	Events/10^5 Flight Hours				
		Surface Damage	Strctr'l Damage	Strctr'l Deform.	Strctr'l Deter.	Total
Personnel Doors						
Canopy Door	Vibration		50			
	Rotor Downwash			25		
	Crew Damage		50	25		
Cabin Door	Vibration		132			
	Fluctuating Loads		240			
	Maintenance/Handling		240			
Cargo Door	Fluctuating Loads		170			
	Maintenance Handling	942	1686	149		
Crew Door	Vibration		27			
	Fluctuating Loads		200			
	Rotor Downwash		250	68		
	Maintenance/Handling		209			
Passenger Door	Vibration		15			
	Crew Damage		15			
Access Doors, Panels and Covers						
Aft Pylon Doors	Fluctuating Loads	77	929			
	Maintenance/Handling	19	319			
Ammo Compartment Doors	Mechanical Shock		22			
	Crew Damage		21			

TABLE A-1 (Continued)

Structures Class/Component	Environment	Events/10^5 Flight Hours				
		Surface Damage	Strctr'l Damage	Strctr'l Deform.	Strctr'l Deter.	Total
Access Panels	Vibration		60			
	Mechanical Shock		99			
	Fluctuating Loads		154			
	Maintenance/Handling		154			
Covers	Fluctuating Loads		1084			
	Maintenance/Handling		271			
Cabin Crown Cover	Fluctuating Loads	230	8165			
	Maintenance/Handling	58	2041			
<u>Fairing and Cowling</u>						
Fairing and Cowling	Fluctuating Loads	58	665			
	Maintenance/Handling	14	160			
	Foreign Object Damage		26			
Power Plant Cowling	Fluctuating Loads		1880			
Forward Pylon Fairing	Fluctuating Loads		1084			
	Maintenance/Handling		271			
Horseshoe Cowling	Fluctuating Loads	135	579			
	Maintenance/Handling	34	145			
Fuel Pod Fairing	Fluctuating Loads		467			
	Maintenance/Handling		111			

TABLE A-1 (Concluded)

Structures Class/Component	Environment	Events/10^5 Flight Hours				
		Surface Damage	Strctr'l Damage	Strctr'l Deform.	Strctr'l Deter.	Total
Firewalls and Baffles	Vibration		458			
	Rotor Downwash		390			
	Maintenance/Handling		300			
<u>Work Platforms</u>						
Pylon Work Platform	Fluctuating Loads	348	756			
	Rotor Cir. Sand and Dust		116			
	Maintenance/Handling	87	189			
Engine Work Platform	Fluctuating Loads		174			
	Maintenance/Handling		43			
Fuselage Work Platform	Fluctuating Loads	58	3415			
	Maintenance/Handling	14	854			
Sponson Work Platform	Maintenance/Handling		21			

APPENDIX B

DAMAGE TOLERANCE AND REPAIRABILITY TEST RESULTS

TABLE B-1. MONOLITHIC PANEL IMPACT TEST RESULTS

Thickness/ Specimen No.	Material	Impact Energy (in-lb)	Damage Depth (inch)	Damage Description
.020-1	Fiberglass/ Epoxy	20	-	Resin fracture bottom ply - .12 dia.
-2		30	-	Fractured bottom ply - .12 dia.
-3		40	.075	Fractured both plies - .625 cross
-4		50	.140	Fractured both plies - .625 dia.
.040-1		20	-	Resin fracture bottom 2 plies - .06 dia.
-2		30	-	Resin fracture bottom 2 plies - .12 dia.
-3		40	-	Resin fracture bottom 3 plies - .12 dia.
-4		50	.014	Fractured all plies - .38 cross
.060-1		20	-	Resin fracture bottom ply - .12 dia.
-2		30	-	Fractured bottom plies, .12 x .06
-3		40	-	Fractured bottom plies, .25 x .12
-4		50	-	Fractured bottom plies, .31 x .18
.080-1		20	-	Resin fracture bottom ply - .12 dia.
-2		30	-	Resin fracture bottom ply - .18 dia.
-3		40	-	Resin fracture bottom ply - .18 dia.
-4		50	-	Fractured bottom ply - .18 dia.
.024-1	Graphite/ Epoxy	20	.014	Fracture top ply - 1 in. lg.
-2		30	.019	Fracture top ply - 1 in. lg.
-3		40	.076	Fracture all plies - .5 x 1.0 in.
-4		50	thru	Panel split into 3 pieces
.040-1		20	-	Slight bulge - bottom ply
-2		30	.007	Fracture bottom ply - 1 in. lg.
-3		40	.035	Fractured all plies - 4 in. crack
-4		50	.048	Fractured all plies - .50 dia. top face
.056-1		20	-	Slight fracture bottom ply - .5 in. lg.
-2		30	-	Slight fracture bottom ply - .75 in. lg.
-3		40	.002	Slight bulge bottom ply - .50 in. lg.
-4		50	.005	Fracture bottom ply - .50 in. lg.
.072-1		20	-	Negligible crack bottom ply - 1 in. lg.
-2		30	-	Negligible crack bottom ply - 1 in. lg.
-3		40	-	Slight bulge bottom ply - 1 in. lg.
-4		50	-	Slight bulge bottom ply - 1 in. lg.
.020-1	Kevlar/ Epoxy	20	-	Resin fracture bottom ply .06 dia.
-2		30	.058	Fractured all plies, .75 in. cross
-3		40	.072	Fractured all plies, .50 in. cross
-4		50	.140	Fractured all plies, .87 in. cross

TABLE B-1 (Concluded)

Thickness/ Specimen No.	Material	Impact Energy (in-lb)	Damage Depth (inch)	Damage Description
.040-1	Kevlar/ Epoxy (cont'd)	20	-	Resin fracture bottom ply .25 in dia.
-2		30	.019	Fractured all plies .50 in. cross
-3		40	.028	Fractured all plies .50 in. cross
-4		50	.043	Fractured all plies .75 in. cross
.060-1		20	-	Resin fracture bottom ply .06 dia.
-2		30	-	Resin fracture bottom ply .31 x .62
-3		40	.006	Resin fracture bottom ply
-4		50	.007	Resin fracture bottom ply .50 cross
.080-1		20	-	No damage
-2		30	-	Resin fracture bottom plies .06 x .25
-3		40	-	Resin fracture bottom plies .38 cross
-4		50	-	Resin fracture bottom plies .50 cross
.016-1	Aluminum	20	.030	Dent, .25 dia.
-2		30	.046	Dent, .38 dia.
-3		40	.058	Dent, .43 dia.
-4		50	.070	Dent, .50 dia.
.025-1		20	.022	Dent, .18 dia.
-2		30	.029	Dent, .25 dia.
-3		40	.050	Dent, .31 dia.
-4		50	.064	Dent, .38 dia.
.032-1		20	.028	Dent, .06 dia.
-2		30	.032	Dent, .12 dia.
-3		40	.045	Dent, .18 dia.
-4		50	.050	Dent, .25 dia.
.040-1		20	.015	Dent, .06 dia.
-2		30	.020	Dent, .12 dia.
-3		40	.025	Dent, .18 dia.
-4		50	.046	Dent, .25 dia.

TABLE B-2. RESULTS OF MONOLITHIC PANEL DAMAGE, REPAIR AND TENSILE TESTS

Specimen Number	Material/ Thickness	Specimen Type	Failure Load (lb)	Location of Failure
-1	Fiberglass/ Epoxy .040 Inch	Control	4180	Gage Section
-2		Control	4425	End of Doubler
-3		Control	4180	Gage Section
-4		Damaged	2105	Through Hole
-5		Damaged	2200	Through Hole
-6		Damaged	2280	Through Hole
-7		Repaired	3470	Through Patch
-8		Repaired	3780	1/2 inch beyond patch
-9		Repaired	3680	1 inch beyond patch
-1	Kevlar/ Epoxy .040 Inch	Control	4220	End of Doubler
-2		Control	5370	End of Doubler
-3		Control	4925	End of Doubler
-4		Damaged	2525	Through Hole
-5		Damaged	2240	Through Hole
-6		Damaged	2450	Through Hole
-7		Repaired	3060	Edge of Patch
-8		Repaired	3960	Edge of Patch
-9		Repaired	3880	Edge of Patch
-1	Graphite/ Epoxy .040 Inch	Control	4180	Under Doubler
-2		Control	7905	End of Doubler
-3		Control	6450	Under Doubler
-4		Damaged	3250	Through Hole
-5		Damaged	4680	Through Hole
-6		Damaged	3450	Through Hole
-7		Repaired	5650	Patch Bond Failure
-8		Repaired	3900	Patch Bond Failure
-9		Repaired	5750	Patch Bond Failure

TABLE B-3. RESULTS OF FIBERGLASS MONOLITHIC PANEL 60-INCH-POUND IMPACT AND TENSILE TEST

Thickness/ Specimen No.	Specimen Type	Failure Load (lb)	Location of Failure
.080-1	Control	6710	End of doubler
-2	Damaged	6810	.25 in from doubler
.060-1	Control	5180	End of doubler
-2	Damaged	4680	Within gage section
.040-1	Control	3600	Within gage section
-2	Damaged	2890	Within gage section
.020-1	Control	1750	.50 in. from doubler
-2	Damaged	1566	1.0 in. from doubler

TABLE B-4. SANDWICH PANEL IMPACT TEST RESULTS

Thickness/ Specimen No.	Material	Impact Energy (in-lb)	Damage Depth (inch)	*Damage Description
.020-1	Fiberglass Facing/ Aluminum Core	20	.056	Dent, .80 in.dia.
-2		30	.072	Dent, .75 in.dia.
-3		40	.125	Fracture in form of a cross .62 dia.
-4		50	.165	Fracture in form of a cross .75 dia.
.030-1		20	.020	Dent, .312 in.dia.
-2		30	.050	Dent, .500 in.dia.
-3		40	.070	Fracture in form of a cross .50 dia.
-4		50	.080	Fracture in form of a cross .62 dia.
.040-1		20	.023	Dent, .12 in.dia.
-2		30	.037	Dent, .25 in.dia.
-3		40	.042	Dent, .312 in.dia.
-4		50	.047	Dent, .38 in.dia.
.020-1	Fiberglass Facing/ Nomex Core	20	.020	Dent, .12 in.dia.
-2		30	.027	Dent, .18 in.dia.
-3		40	.064	Fracture .38 dia.
-4		50	.088	Fracture in form of a cross .62 dia.
.030-1		20	.005	Resin fracture (slight dent) .12 in.dia.
-2		30	.012	Resin fracture (slight dent) .18 in.dia.
-3		40	.042	Fracture in form of a cross .50 in.dia.
-4		50	.064	Fracture in form of a cross .38 in.dia.
.040-1		20	.004	Resin fracture .12 in.dia.
-2		30	.006	Resin fracture .18 in.dia.
-3		40	.008	Resin fracture .18 in.dia.
-4		50	.010	Resin fracture .25 in.dia.
.024-1	Graphite Facings/ Aluminum Core	20	.038	Dent .38 in.dia.
-2		30	.100	Fracture .80 in.dia.
-3		40	.100	
-4		50	.112	Fracture .50 in x .38 in.
.032-1		20	.024	Dent, .25 in.dia.
-2		30	.060	Fracture .31 in.dia.
-3		40	.090	Fracture .38 in.dia.
-4		50	.105	Fracture .43 in.dia.
.045-1		20	.010	Dent, .12 in.dia.
-2		30	.042	Dent, .18 in.dia.
-3		40	.052	Fracture .25 in.dia.
-4		50	.058	Fracture .38 in.dia.
.024-1	Graphite Facings/ Nomex Core	20	.012	Dent, .12 in.x .25 in.
-2		30	.028	Crack, 3/4 in.lg.
-3		40	.070	Fracture .38 in.dia.
-4		50	.096	Fracture .25 in.x .50 in.

*All damage in top faces

TABLE B-4 (Continued)

Thickness/ Specimen No.	Material	Impact Energy (in-lb)	Damage Depth (inch)	*Damage Description
.032-1	Graphite Facings/ Nomex Core (Cont'd)	20	.005	Dent, .18 in. dia.
-2		30	.015	Dent, .25 in. dia.
-3		40	.030	Dent, .31 in. dia.
-4		50	.040	Dent, .38 in. dia.
.045-1		20	.003	Slight dent .06 in dia.
-2		30	.007	Dent, .12 in. dia.
-3		40	.029	Dent, .31 in. dia.
-4		50	.030	Dent, .25 in. x .38 in
.020-1	Kevlar Facings/ Aluminum Core	20	.020	Dent, .25 in. dia.
-2		30	.088	Fracture, .38 in. dia.
-3		40	.145	Fracture,.50 in.cross
-4		50	.242	Fracture, .75. in cross
.030-1		20	.070	Dent, .25 in dia.
-2		30	.078	Fracture, .38 in. dia.
-3		40	.105	Fracture, .50 in. cross
-4		50	.180	Fracture, .6 in. cross
.040-1		20	.025	Dent, .25 in dia.
-2		30	.060	Fracture .38 in. cross
-3		40	.084	Fracture .50 in. cross
-4		50	.092	Fracture .50 in. cross
.020-1	Kevlar Facings/ Nomex Core	20	.013	Dent, .12 in. dia.
-2		30	.010	Fracture .50 in. dia.
-3		40	.068	Fracture .62 in. cross
-4		50	.084	Fracture .75 in. cross
.030-1		20	.020	Dent, .18 in. dia.
-2		30	.045	Fracture .43 in. cross
-3		40	.055	Fracture .50 in. cross
-4		50	.078	Fracture .62 in. cross
.040-1		20	.016	Dent, .18 in. dia.
-2		30	.035	Fracture .50 in. cross
-3		40	.054	Fracture .50 in. cross
-4		50	.060	Fracture.62 in. cross
.016-1	Aluminum Facings/ Aluminum Core	20	.044	Dent, .25 in. dia.
-2		30	.064	Dent, .31 in. dia.
-3		40	.074	Dent, .38 in. dia.
-4		50	.090	Dent, .43 in. dia.

*All damage in top faces

TABLE B-4 (Concluded)

Thickness/ Specimen No	Material	Impact Energy (in-lb)	Damage Depth (inch)	*Damage Description
.020-1	Aluminum Facings/ Aluminum Core (Cont'd)	20	.047	Dent, .18 in. dia.
-2		30	.064	Dent, .25 in. dia.
-3		40	.077	Dent, .31 in. dia.
-4		50	.085	Dent, .31 in. dia.
.032-1		20	.042	Dent, .12 in. dia.
-2		30	.045	Dent, .18 in. dia.
-3		40	.044	Dent, .25 in. dia.
-4		50	.065	Dent, .38 in. dia.
.016-1	Aluminum Facings/ Nomex Core	20	.050	Dent, .18 in. dia.
-2		30	.066	Dent, .25 in. dia.
-3		40	.079	Dent, .25 in. dia.
-4		50	.089	Dent, .38 in. dia.
.020-1		20	.034	Dent, .18 dia.
-2		30	.055	Dent, .18 dia.
-3		40	.074	Dent, .25 in. dia.
-4		50	.080	Dent, .25 in. dia.
.032-1		20	.026	Dent, .12 in. dia.
-2		30	.032	Dent, .18 in. dia.
-3		40	.046	Dent, .18 in. dia.
-4		50	.050	Dent, .25 in. dia.

*All damage in top faces

TABLE B-5. RESULTS OF SANDWICH PANEL IMPACT AND BEAM SHEAR TESTS

Specimen Thickness/ Number	Material	Specimen Type	Failure Load (lb)	Remarks
.020-1	Fiberglass Facings/ Nomex Core	Control	264	Top facing fracture & core crush across load point.
-2		Damaged	94	Top facing delamination & core crush.
-3		Repaired	300	Core shear at end of patch.
.030-1		Control	332	Bottom facing fracture at load point - Core crush between reaction points.
-2		Damaged	157	Top facing fracture, core crush across load point.
-3		Repaired	350	Core shear at end of patch.
.040-1		Control	345	Core crush across load point.
-2		Damaged	225	Bottom facing break at one reaction point and core crush.
-3		Repaired	255	Bottom face and core fracture at load point.
.020-1	Fiberglass Facing/ Aluminum Core	Control	318	Core shear.
-2		Damaged	122	Top face and core crush at load point.
-3		Repaired	215	Upper face fracture and core failure at end of patch.
.030-1		Control	321	Bottom face and core crush at one reaction point.
-2		Damaged	205	Top face and core crush at load point.
-3		Repaired	340	Total fracture at load point.
.040-1		Control	356	Bottom face and core crush between reaction points.
-2		Damaged	262	Bottom face and core crush between reaction points.
-3		Repaired	390	Bottom face and core fracture at end of patch.
.024-1	Graphite Facings/ Nomex Core	Control	417	Top face and core fracture between load points.
-2		Damaged	175	Top face and core fracture between load points.
-3		Repaired	645	Upper face fracture at end of patch and core crush under patch.
.032-1		Control	634	Top face and core fracture between load points.
-2		Damaged	250	Top face and core fracture between load points.
-3		Repaired	660	Bottom face fracture at end of patch & core crush under patch.
.040-1		Control	674	*Bottom face & core failure between load points.
-2		Damaged	416	Top face & core failure between load points.
-3		Repaired	642	Upper face fracture at end of patch and core crush under patch.

*Also core separation from facing

TABLE B-5 (Continued)

Specimen Thickness/ Number	Material	Specimen Type	Failure Load (lb)	Remarks
.024-1	Graphite Facings/ Aluminum Core	Control	623	*Bottom face and core failure between load points.
-2		Damaged	346	Top face and core failure between load points.
-3		Repaired	710	Bottom face and core fracture at end of patch.
.032-1		Control	683	*Bottom face and core failure between load points.
-2		Damaged	356	Top face and core failure between load points.
-3		Repaired	765	Bottom face and core fracture at end of patch.
.040-1		Control	644	Top face and core failure between load points.
-2		Damaged	514	Top face and core failure between load points.
-3		Repaired	747	Bottom face and core fracture at end of patch.
.020-1	Kevlar Facings/ Nomex Core	Control	123	Top facing crease and core failure at center.
-2		Damaged	64	Top facing crease and core failure at center.
-3		Repaired	149	Upper face and core fracture at end of patch.
.030-1		Control	176	Core shear.
-2		Damaged	94	Top facing crease and core failure at center.
-3		Repaired	198	Upper face and core fracture at end of patch.
.040-1		Control	236	Core shear.
-2		Damaged	169	Top facing crease and core failure at center.
-3		Repaired	258	Upper face and core fracture at end of patch.
.020-1	Kevlar Facings/ Aluminum Core	Control	144	Core shear.
-2		Damaged	64	Top facing crease and core failure at center.
-3		Repaired	160	Upper face and core fracture at end of patch.
.030-1		Control	192	Top facing crease and core failure at center.
-2		Damaged	108	Top facing crease and core failure at center.
-3		Repaired	200	Upper face and core fracture at end of patch.
.040-1		Control	266	Core shear.
-2		Damaged	157	Top facing crease and core failure at center.
-3		Repaired	320	Core shear - No facing failures.

*Also core separation from facing

TABLE B-5 (Concluded)

Specimen Thickness/ Number	Material	Specimen Type	Failure Load (lb)	Remarks
.016-1	Aluminum Facings/ Nomex Core	Control	175	Top face buckle and core failure at one load point.
-2		Damaged	175	Top face crease and core failure between load points.
-3		Repaired	275	Beam bending - stopped at 275 lb.
.020-1		Control	307	Top face buckle and core failure between load points.
-2		Damaged	131	Top face buckle and core failure between load points.
-3		Repaired	350	Top face crease and core failure at patch end.
.032-1		Control	492	Top face crease and core failure at one load point.
-2		Damaged	304	Top face crease and core buckle between load point.
-3		Repaired	545	Top face crease and core failure at patch.
.016-1	Aluminum Facings/ Aluminum Core	Control	267	Core shear.
-2		Damaged	154	Top face crease and core failure between load points.
-3		Repaired	300	Beam bending - stopped at 300 lbs.
.020-1		Control	298	Core shear.
-2		Damaged	199	Top face crease and core failure between load point.
-3		Repaired	358	Bottom face and core failure at patch.
.032-1		Control	462	Top face crease and core failure at one load point.
-2		Damaged	330	Top face crease and core failure between load points.
-3		Repaired	575	Top face and core failure 1 in. before end of patch.

*Also core separation from facing

APPENDIX C

DETAILED R&M ANALYSIS

OF A KEVLAR SKIN-SKELETON REAR FUSELAGE DESIGN

WORKSHEET #R1

DAMAGE POTENTIAL ASSESSMENT

Structure: Rear Fuselage Kevlar Skin - Skeleton
Application: Utility aircraft; combat environment

Environmental Hazard	Hazard Frequency	Hazard Exposure	Damage Potential
Vibration	High	Low	Moder.
Airborne Particles/F.O.D.	Low	Neg.	--
Foot Traffic	High	Low	Moder.
Dropped Tools/Parts	Moder.	Low	Low
Dropped/Shifting Cargo/Stores	Moder.	Moder.	Moder.
Door Slamming	Low	N/A	--
Rough Handling	Low	N/A	--
Bird Strikes	Low	Neg.	--
Impact with Terrain Objects	Moder.	High	High
Work Stands/ Ground Vehicles	Low	Low	--
Ballistic Impacts	Moder.	Moder.	Moder.
Corrosive Elements	Moder.	High	High

Rate for Type Aircraft,
Mission & Environment
(See Guide # G1)

Rate for Type Structure,
Location & Protection
(See Guide # G2)

Worksheet #R2

DAMAGE TOLERANCE ASSESSMENT

Structure: Rear Fuselage; Kevlar Skin - Skeleton

		Abrasion/ Chafing	Denting	Puncture	Delamination	Cracking	Fastener Damage	Crushing	Buckling	Corrosion
	Potential Damage Modes	Abrasion/ Chafing	Denting	Puncture	Delamination	Cracking	Fastener Damage	Crushing	Buckling	Corrosion
	Material Damage Tolerance (See Guide)	Low	High	Low	Low	Moder.	Low	Low	Low	N/A
	Design Damage Tolerance (Predominant Attributes)									
X	Accessible to Inspection	⊕	⊕	⊕	⊕	⊕	⊕			⊕
X	Heavily Loaded	▨	⊕	⊕	⊕	⊕	⊕	⊕	⊕	
X	Few Interface Constraints	▨	▨	▨	⊕	⊕	▨	▨	▨	▨
X	Monolithic or Stiffened Construction	▨	⊕	⊕	⊕	▨	▨	⊕	▨	▨
	Sandwich Construction	▨	▨	▨	▨	+	▨	▨	+	▨
X	Closed Section Stiffeners	▨	▨	▨	▨	▨	▨	▨	⊕	▨
X	Bonded or Co-Cured Assembly	▨	▨	▨	▨	⊕	⊕	▨	▨	▨
	Damage Tolerance Rating	Moder.	High	Moder.	Moder.	High	High	Moder.	Moder.	–

Worksheet #R3
DAMAGE MODE ASSESSMENT

Structure: Rear Fuselage; Kevlar Skin - Skeleton

Environmental Hazard	Damage Potential	Abrasion/ Chafing	Denting	Puncture	Delamination	Cracking	Fastener Damage	Crushing	Buckling	Corrosion
Damage Tolerance Rating from Worksheet #R2 →		Moder.	High	Moder.	Moder.	High	High	Moder.	Moder.	—
Damage Potential Rating from Worksheet #R1 →										
Vibration	Moder.	Moder.	▨	▨	Moder.	Low	Low	▨	▨	▨
Airborne Particles/ F.O.D.	—						▨	▨	▨	▨
Foot Traffic	Moder.	Moder.*	Low	▨	Moder.*	Low	▨	Moder.*	Moder.*	▨
Dropped Tools/ Parts	Low	▨		Low	Low		▨	Low	Low	▨
Dropped/Shifting Cargo/Stores	Moder.	Moder.	Low	Moder.*	Moder.*	Low	▨	Moder.*	Moder.*	▨
Door Slamming	—	▨	▨	▨				▨		▨
Rough Handling	—	▨								▨
Bird Strikes	—	▨					▨			▨
Impact with Terrain Objects	High	▨	Moder.	High*	High*	Moder.	▨	High*	High*	▨
Work Stands/ Ground Vehicles	—	▨					▨			▨
Ballistic Impacts	Moder.	▨	▨	Moder.*	Moder.*	Low	▨	▨	▨	▨
Corrosives	High	▨	▨	▨	▨	▨	▨	▨	▨	
Damage Mode Assessment →		Moder.	Low	Moder.	Moder.	Low	Low	Moder.	Moder.	—

* Indicates areas of concern noted in the R&M Assessment Summary

WORKSHEET #M1
REPAIRABILITY ASSESSMENT

Structure: Rear Fuselage; Kevlar Skin - Skeleton

		Types of Repair			
	Factor	Standard Field Repair	Complex Repair	Custom-Engineered Repair	No Repair
Design Related	Load Intensity	☒ Light to Moderate	☐ Moderate to Heavy	☐ Heavy	☐ Heavy
	Shape/Contour	☐ Flat/Single Curvature	☒ Single/Double Curvature		☐ Complex Shape/Contour/Buildup
	Interface Constraints	☒ Few	☐ Some	☐ Many	
	Skin/Web Form	☐ Monolithic Sandwich	☒ Integrally Stiffened Sheet		
	Stiffener/Frame Form	☐ Open Section	☒ Closed Section		
Maintenance Related	Repair Materials	☒ Stock/Bulk Items	☐ Special Kits	☐ Special Storage/Handling	
	Environmental Requirements	☒ Field Environment	☐ Controlled Environment	☐ Clean Room Conditions	
	Tools and Equipment	☒ Standard Field Type	☐ Special Field Type	☐ Factory Type	
	Personnel Skills	☒ Low Skill Level	☐ Intermediate Skill Level	☐ High Skill Level	

Typical Component	Aircraft Skin/Fairing	Intermediate Frame	Longeron	Transmission Support Beam

This Structure Will Require Primarily →	☒ Standard Field Repair	☐ Complex Repair	☐ Custom Engineered Repair	☐ Non-Repairable

WORKSHEET #M2
REPLACEABILITY ASSESSMENT

Structure: Rear Fuselage; Kevlar Skin - Skeleton

Factor	Simple Field Replacement	Complex Field Replacement	Depot Replacement	No Replacement
Type of Joint	☐ Simple Bolted Joint	☒ Semi-Permanent Fasteners	☐ Custom Fitted/ Shimmed	☐ Integral Molded/ Machined Structure
Obstructions and Interfaces	☐ Minor Parts and Components	☐ Major Components	☒ Major Components/ Plumbing/ Wiring	
Jigs and Fixtures	☐ None	☐ Field Type	☒ Factory Type	
Spares	☐ Small/ Inexpensive	☐ Large/ Inexpensive	☒ Large/ Expensive	
Aircraft Downtime	☐ Low	☐ Moderate	☒ Extensive	

Typical Component	Fairing/ Door	Tail Cone	Rear Fuselage	Transmission Support Beam

Structure is →	☐ Simple Field Replacement	☐ Complex Field Replacement	☒ Depot Replacement	☐ Non-Replaceable

WORKSHEET #M3
MAINTAINABILITY ASSESSMENT

Structure: Rear Fuselage; Kevlar Skin - Skeleton

Factor	Good	Fair	Poor
Accessibility	Both Sides ☒	One Side ☐	Obstructed/ Inaccessible ☐
Inspectability	Visual ☒	Portable NDT ☐	Shop NDT ☐
Repairability (Worksheet #M1)	Standard Field Repair ☒	Complex Repair ☐	Custom-Engineered Repair ☐
Level of Repair	On Aircraft ☒	Field Shop ☐	Depot ☐
Replaceability (Worksheet #M2)	Easy Field Replacement ☐	Difficult Field Replacement ☐	Depot Replacement ☒
Expendability	Low Cost ☐	Moderate Cost ☒	High Cost ☐

Overall Maintainability Rating →	Good ☒	Fair ☐	Poor ☐

WORKSHEET # RM1
R&M ASSESSMENT SUMMARY

Structure:
Rear Fuselage; Kevlar Skin - Skeleton

Structural Reliability: Good

Hardware Reliability: Good

Maintainability: Good*

Overall: Good

Type and Location of Damage	Source	Expected Frequency	R&M Concern
Punctures, delamination, crushing and buckling - exterior of tub section	Landing on obstructions, striking terrain objects during NOE flying.	Low	Moderate concern. Damage could be extensive, but should be infrequent.
Punctures, delamination, crushing and buckling - interior of stowage area.	Throwing equipment into stowage area.	Moderate	Moderate concern. Damage should be light, but not always visible.
Abrasion, delamination, crushing, buckling of structure adjacent to fuselage steps.	Foot traffic.	Moderate	Minor concern. Most damage superficial and obvious. Could be maintenance nuisance.
Punctures, delamination - tub section	Ballistic impacts.	Low	Moderate concern. Large caliber weapons will cause extensive damage. Quick-fix and large area repair techniques require further development.

* Based on the assumption that effective field repair techniques are developed.